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V/STOL TILT ROTOR AIRCRAFT STUDY

VOLUME VII

TILT ROTOR FLIGHT CONTROL PROGRAM FEEDBACK STUDIES

H. R. Alexander

W. Eason

K. Gillmore

J. Morris

R. Spittle

MARCH 1973

NASA CR 114 600

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D222-10060-3

THE ***BOEING*** COMPANY

VERTOL DIVISION · PHILADELPHIA, PENNSYLVANIA

CODE IDENT. NO. 77272

NUMBER ___D222-10060-3___

TITLE ___V/STOL TILT ROTOR AIRCRAFT STUDY -___

___TILT ROTOR FEEDBACK CONTROL TECHNOLOGY___

ORIGINAL RELEASE DATE_____. FOR THE RELEASE DATE OF
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MODEL ___NA___ CONTRACT ___NAS2-6598 Phase III___ Task C

ISSUE NO. _____ ISSUED TO: _____

PREPARED BY	H. Alexander	DATE	3/26/73
APPROVED BY	K. Gillmore	DATE	4/3/73
APPROVED BY	D. Richardson	DATE	4/4/73
APPROVED BY	W. Peck	DATE	4/3/73

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FORM 46281 (3/67)

SHEET   II

ACTIVE SHEET RECORD

SHEET NUMBER	REV LTR	ADDED SHEETS					SHEET NUMBER	REV LTR	ADDED SHEETS				
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123								163	A						
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SHEET V

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223						263					
224						264					
225						265	A				
226						266					
227						267					
228						268	A				
229						269	A				
230						270					
231						271					
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286	A							325							
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290	A							329							
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REVISIONS			
LTR	DESCRIPTION	DATE	APPROVAL
A	Issued to incorporate comments of the customer, National Aeronautics and Space Administration, Moffett Field, California	7-11-73	*Hnu*

FOREWORD

This report is one of a series prepared by The Boeing Vertol Company, Philadelphia, Pennsylvania for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California under contract NAS2-6598. The studies reported under Volumes I through IV and VIII through X were jointly funded by NASA and the U. S. Army Air Mobility Research and Development Laborator   s Directorate. Volumes V through VII were funded by   Air Force Flight Dynamics Laboratory, Wright Patterson Ai)   se, Ohio.

This contract was adm   a by the National Aeronautics and Space Administration.   .ichard J. Abbott was the Contract Administr   , Mr. Gary B. Churchill, Tilt Rotor Research Aircraft Project Office, was the Technical Monitor, and coordination and liaison with the U. S. Air Force Flight Dynamics Laboratory was through Mr. D. Fraga. The Boeing-Vertol Project Engineer for the work reported in Volume VII was Mr. H. R. Alexander.

The complete list of reports published under this contract is as follows:

ABSTRACT

An exploratory study has been made of the use of feedback control in tilt rotor aircraft. This has included the use of swashplate cyclic and collective controls and direct lift control.

Various sensor and feedback systems are evaluated in relation to blade loads alleviation, improvement in flying qualities, and modal suppression.

Recommendations are made regarding additional analytical and wind tunnel investigations and development of feedback systems in the full scale flight vehicle. Estimated costs and schedules are given.

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SYMBOLS

A_1	Cyclic pitch coefficient of cos term
a_1, a	Gain in A_1 cyclic channel
B_1	Cyclic pitch coefficient of sin term
b_1, b	Gain in B_1 cyclic channel
C_{m_α}	Rotor pitching moment coefficient due to angle of attack
C_{N_α}	Rotor normal force coefficient due to angle of attack
C_{x_α}	Rotor yawing moment coefficient due to angle of attack
C_{Y_α}	Rotor side force coefficient due to angle of attack
DLC	Direct lift control
F_{xA_1}	Normal force due to A_1 cyclic
F_{x_α}	Rotor normal force due to angle of attack
F_{yA_1}	Rotor side force due to A_1 cyclic
F_{Y_α}	Rotor side force due to angle of attack
G_{GOV}	Governor gain, governor proportional gain
G_p	Gain in lo-rate pitch channel - 26 ft rotor test
G_y	Gain in lo-rate yaw channel - 26 ft rotor test
K_β	Flapping stiffness in simplified rotor math model
K_ζ	Lagging stiffness in simplified rotor math model
K	Gain, governor integral gain
M_{xA_1}	Rotor yawing moment due to A_1 cyclic
M_{yA_1}	Rotor pitching moment due to A_1 cyclic
M_{y_α}	Rotor pitching moment due to angle of attack

XXXI

SYMBOLS

M_{x_α}	Rotor yawing moment due to angle of attack
M_u	Variation in aircraft pitching moment with velocity
NR1	Lagging frequency ratio for rotor blade ω_ℓ/Ω
NR2	Flapping frequency ratio for rotor blade ω_f/Ω
1P	One per rev
S	Laplace operator
T_u	Variation in thrust with velocity
V	Aircraft velocity
W	Aircraft weight
ω	Frequency RAD/SEC
ω_β	Flap frequency
ω_c	Wing chord frequency
ω_L	Lag frequency
ω_N	Natural frequency
ω_α	Wing torsion frequency
ω_V	Wing vertical bending frequency
$\dot{z}_{VB}$	Vertical velocity of wing tip
$\ddot{z}_{VB}$	Vertical acceleration of wing tip
z_{VB}	Vertical translation of wing tip

SYMBOLS

α	Angle of attack
γ	Locke number
δ_E	Elevator deflection, positive down
δ_F	Flap deflection, positive down
δ_S	Spoiler deflection, positive up
ζ	Damping ratio
θ_0	Rotor blade collective pitch angle
λ	Inflow ratio
τ	Control system time constant
ψ	Azimuthal location

1. SUMMARY

This work was carried out under Task C of NASA-Army Contract NAS2-6598.

The subject of this research contract was the application of feedback control in tilt rotor aircraft to accomplish the following objectives:

- Improve handling qualities

- Reduce blade loads

- Alleviate gust response

- Modal suppression and augmentation of aeroelastic stability

The feedback systems are primarily concerned with the application of swashplate cyclic and collective feedback, although a study has also been made of the use of flaps, spoilers and elevators for direct lift control. This work is preceded by the results of a parametric study of rotor derivatives in which special attention is given to the influence of blade frequencies. Load correlation with test data is demonstrated. Important differences are noted in the derivatives with respect to shaft angle of attack (alpha) and cyclic pitch (A_1). The low in-plane stiffness blade is seen to have a negative hub pitching moment under normal flight conditions so that the net destabilizing effect of the rotor is less than would be the case for a high stiffness blade and this is confirmed by test data. This has beneficial effects on the horizontal and vertical

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tail area requirements.

The blade load alleviation and stability augmentation studies show that various alternative objectives may be met using cyclic feedback. Alternative systems designed to minimize flapping and net pitching moment are examined. The requirements for cyclic pitch gain and azimuth in dynamic gust response situations are compared with those suitable for steady state alleviation and are shown to have differences. Methods or reading these differences are identified and recommended for further study. No stability problems were encountered up to optimal values of gain in any of the systems examined, although additional shaping is recommended in one case to increase the phase margin.

Direct lift control of normal acceleration response in turbulent air using flaps and spoilers feedback is examined. Reductions in normal acceleration demonstrated are approximately 45% in cruise and 15% in transition. This is considered highly encouraging since they are obtained without special effort at optimization or the use of elevator feedback to trim pitching moments due to flap and spoiler application.

In a low disk loading tilt rotor configuration, the combined

requirements of rotor governing and gust alleviation are shown to determine the selection of a governing system. A governing system, which senses the RPM variation of each rotor and feeds back to collective pitch an averaged signal, is shown to be optimal. A system design is discussed which meets selected error criteria in the presence of (1-cosine) and random axial gusts. The system is shown to be compatible with the flying quality characteristics of the aircraft. There is no significant degradation in basic flying qualities and at the same time a very substantial attenuation in responsiveness to axial gusts is noted.

The use of high rate swashplate feedback for modal suppression is shown to provide very substantial amounts of additional damping in the selected modes. Predicted results are compared with those obtained during full-scale tests. Attempts to introduce additional damping in blade modes showed promise but an extensive investigation was not made since this topic is currently of academic interest only. General conclusions and suggestions for additional study are made in each section. One general conclusion is that the full potential of some systems depends on the complementary effects from associated systems. These relationships are summarized in Table 9.1. For example, the effectiveness of a blade load alleviation

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system will be augmented if aircraft vertical and pitching response to gusts is reduced by direct lift feedback control. Conversely, the effectiveness of a direct lift system in reducing cabin acceleration will be improved when working in parallel with a high rate swashplate modal suppression system. Thus, there is a strong case for eventually implementing all the systems discussed, although priority should be given to conventional SAS, rotor governing and load alleviation systems.

The implications of these proposals on hardware is discussed in Section 10. This section also considers the implications of fail-safe and fail-functional requirements and provides schematics for the various systems discussed. The changes necessary to the currently proposed M222 system in order to implement each of these systems are noted. The impact of converting to a fly-by-wire system is discussed.

Section 11 presents a number of recommended programs which are classified under the heading of additional analytical studies, wind tunnel programs and implementation in a full-scale flight program.

Tentative costs and schedules for each of the proposed pro-

grams are given in Section 12.

2. PROP/ROTOR HUB FORCE AND MOMENT DERIVATIVES

2.1 BACKGROUND

Of fundamental importance to any feedback system using cyclic
pitch is the manner in which hub forces and moments respond
to changes in angle of attack and increments of pitch and
yaw cyclic. It is known from previous work that rotor
derivatives are sensitive to a number of parameters, in
particular, blade frequency. In the case of soft in-plane
prop/rotors the derivatives with respect to shaft angle of
attack (alpha derivatives) have been shown to be particularly
sensitive to variations in lead-lag frequency ratios near 1
per rev. An investigation of prop/rotor alpha derivative
sensitivity to blade frequency and lock number is reported in
Reference 2.1. The results reported here add to this body
of knowledge by providing information on the behavior of hub
forces and moments with respect to cyclic pitch (A_1 derivatives).
The mathematical model used for this study is shown in Figure
2.1. This model was chosen because it permits independent
variation of flap and lag frequency without restriction. The
mass and airfoil properties used are those of the Model 222
26-foot diameter prop rotor. Results for the same rotor
with its actual root restraints are given for smaller variations
in blade flap and lag per rev frequency and over a range of
altitude and velocity. Since there are currently in use a
number of different approaches toward non-dimensionalizing
rotor derivatives, the results given in this report are quoted

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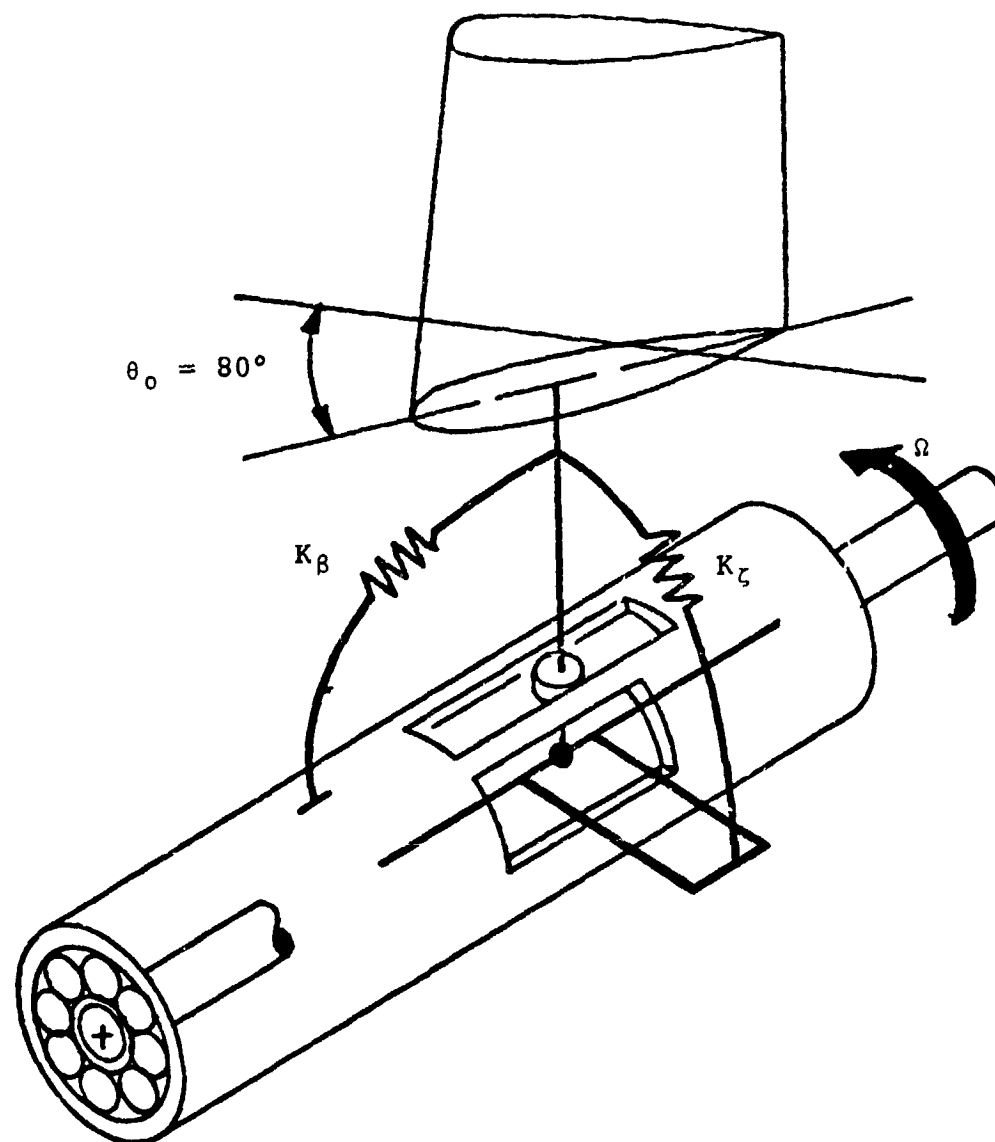
Figure 2.1   Mathematical Model for Rotor Derivative
              Parametric Trend Studies

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in dimensional units to avoid ambiguity. Physical properties of the rotor are given in Appendix A.

2.2 VARIATION OF DERIVATIVES WITH LEAD-LAG FREQUENCY RATIO

The behavior of force and moment derivatives over the range of 0.6 to 2.0 per rev in lead-lag and 1.0, 1.3 and 1.5 for flap is given in Figures 2.2 through 2.5. It is noted that the alpha and cyclic derivatives share the high degree of sensitivity to lead-lag frequency ratio particularly values of N_{R_1} just below 1.0.

In both sets of derivatives normal force and yawing moment attain a minimum at $N_{R_1} = 1$, and that side force and pitching moment change sign when N_{R_1} drops below a value of unity.

This result for the alpha derivatives is the same as for the earlier study published in Reference 2.1; the present study confirms that similar behavior is encountered in the A_1 derivatives.

2.3 VARIATION OF LEAD-LAG AND FLAP FREQUENCY RATIO OVER A LIMITED RANGE

Figures 2.6 through 2.9 present carpets of alpha and cyclic derivatives for the Model 222 26-foot diameter rotor at 100 knots and 385 RPM. Since in the sign convention used positive A_1 means that the blade angle is maximum at azimuth zero (top center), and shaft angle of attack implies maximum excursion in blade angle of attack at 90-degrees, some correspondence is expected between F_{X_α} and $F_{Y_{A_1}}$, M_{Y_α} and $M_{X_{A_1}}$, etc. It is

V = 250 KNOTS
Ω = 386 RPM
λ = .8
γ = 6.6

N_{R_1} = LEAD-LAG FREQUENCY RATIO

N_{R_2} = FLAP FREQUENCY RATIO

$F_{X_{A_1}}$
NORMAL FORCE IN
THOUSANDS OF
LBS/RAD OF A_1

20

1.5

1.3

0

.4   .8   1.2   1.6   2.0

N_{R_1}

1.5
1.0 N_{R_2}
1.3

1.0

-20

N_{R_1}

.4   .8   1.2   1.6   2.0

0

$F_{Y_{A_1}}$
SIDE FORCE IN
THOUSANDS OF
LBS/RAD OF A_1

1.3

-40

1.5

1.0

1.5
1.3 N_{R_2}

1.0

-80
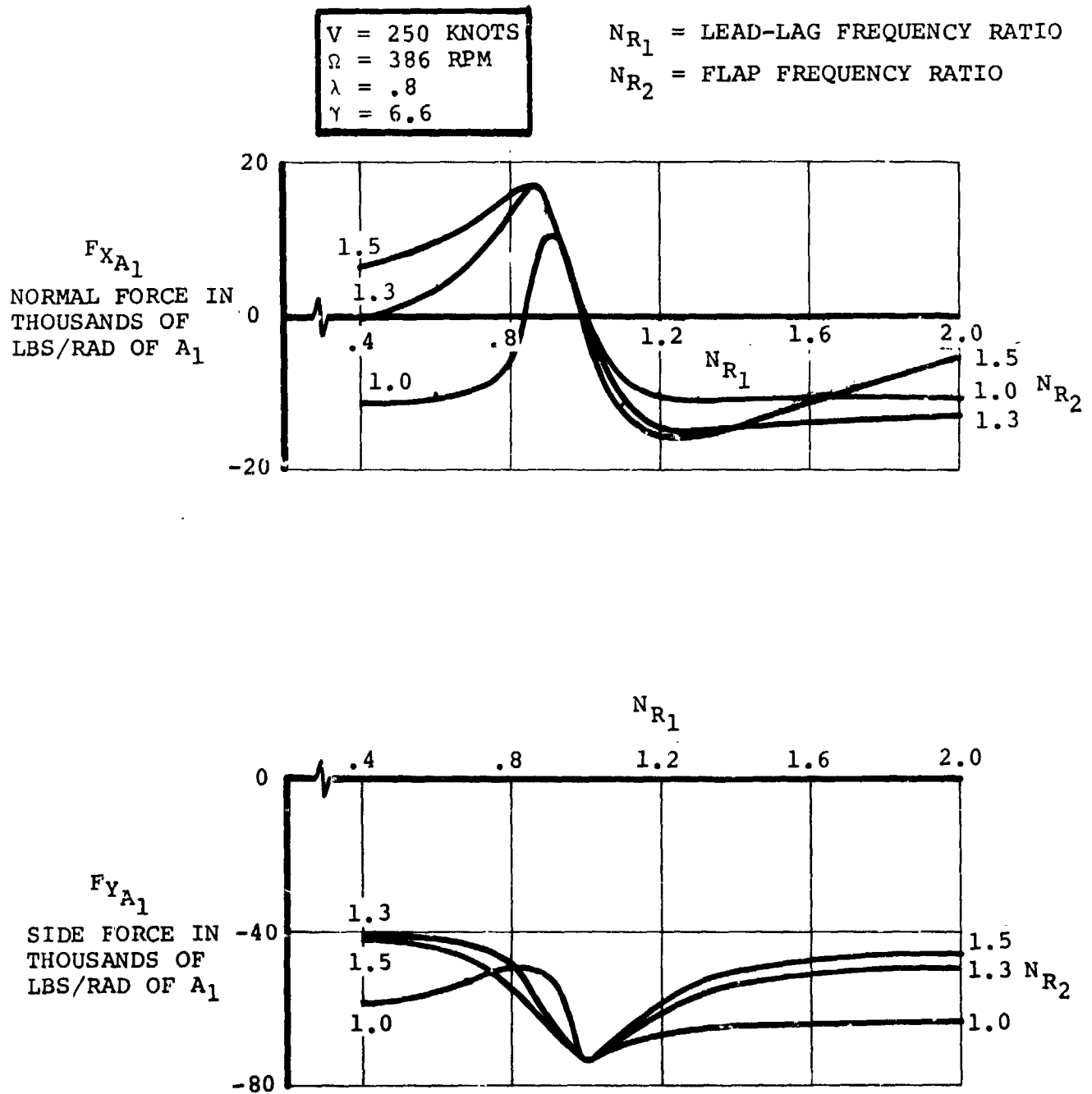
FIGURE 2.2.   EFFECT OF VARIATION OF BLADE FREQUENCY RATIOS
ON CYCLIC FORCE DERIVATIVES – 250 KNOTS

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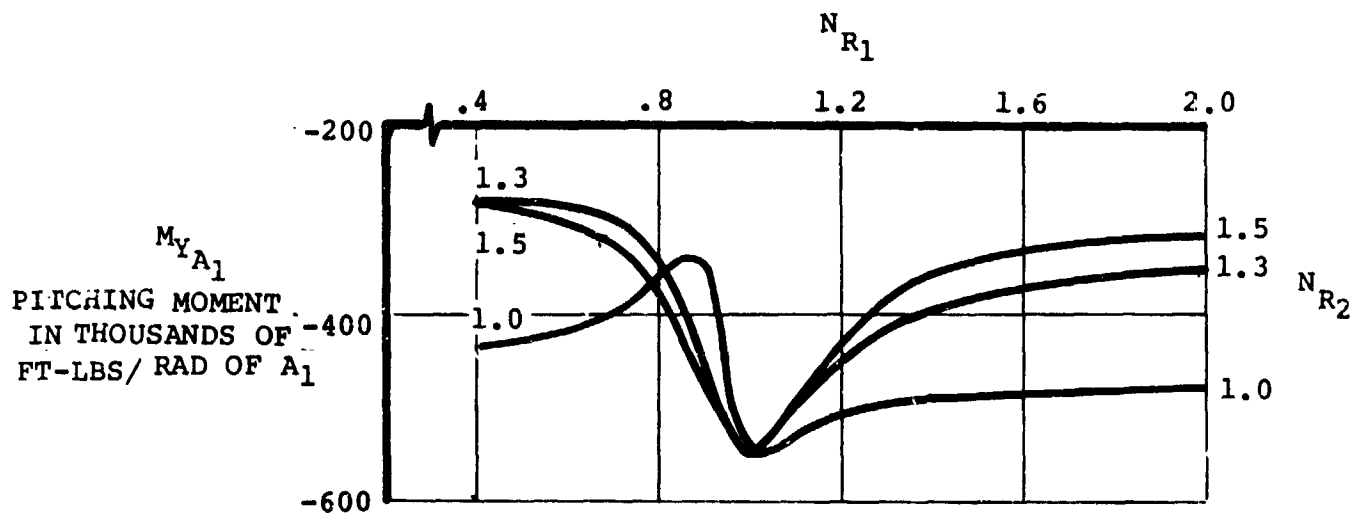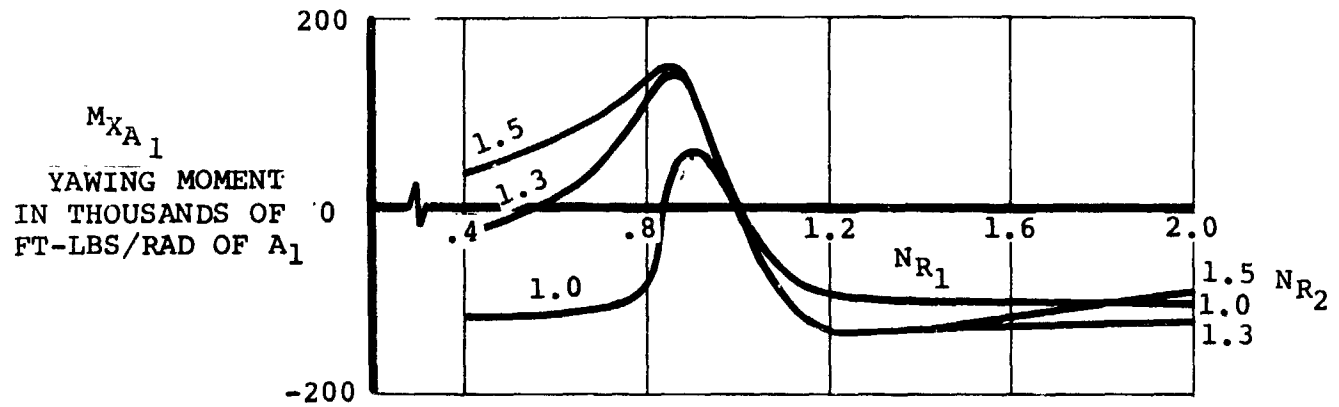
FIGURE 2.3. EFFECT OF VARIATION OF BLADE FREQUENCY RATIOS
ON CYCLIC MOMENT DERIVATIVES - 250 KNOTS
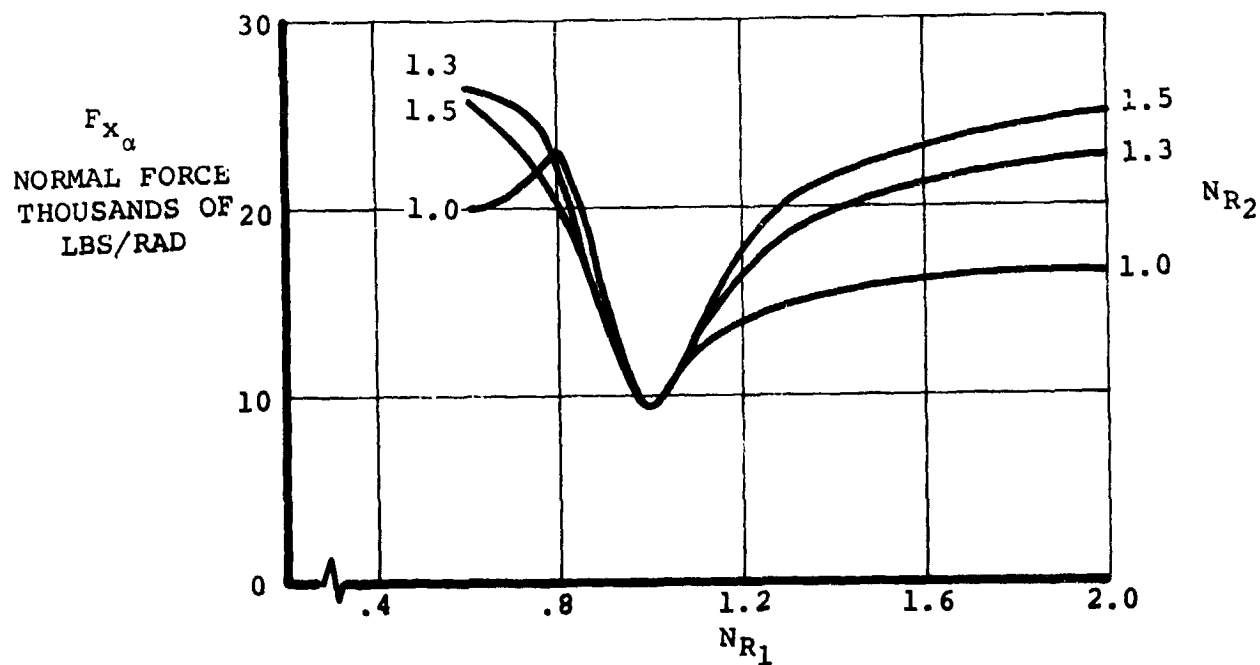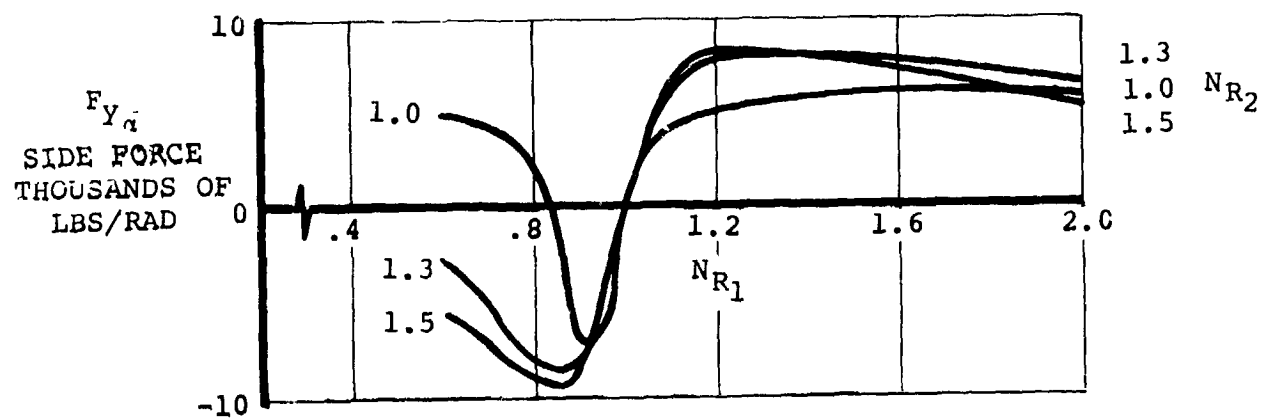
FIGURE 2.4. VARIATION OF PROP/ROTOR NORMAL AND
SIDE FORCE ALPHA DERIVATIVES WITH
RESPECT TO BLADE LEAD-LAG FREQUENCY.

11

$$
\begin{array}{l}
V = 250 \ \text{KNOTS} \\
\Omega = 386 \ \text{RPM} \\
\lambda = .8 \\
\gamma = 6.6
\end{array}
$$

N_{R_1} = LEAD-LAG FREQUENCY RATIO

N_{R_2} = FLAP FREQUENCY RATIO
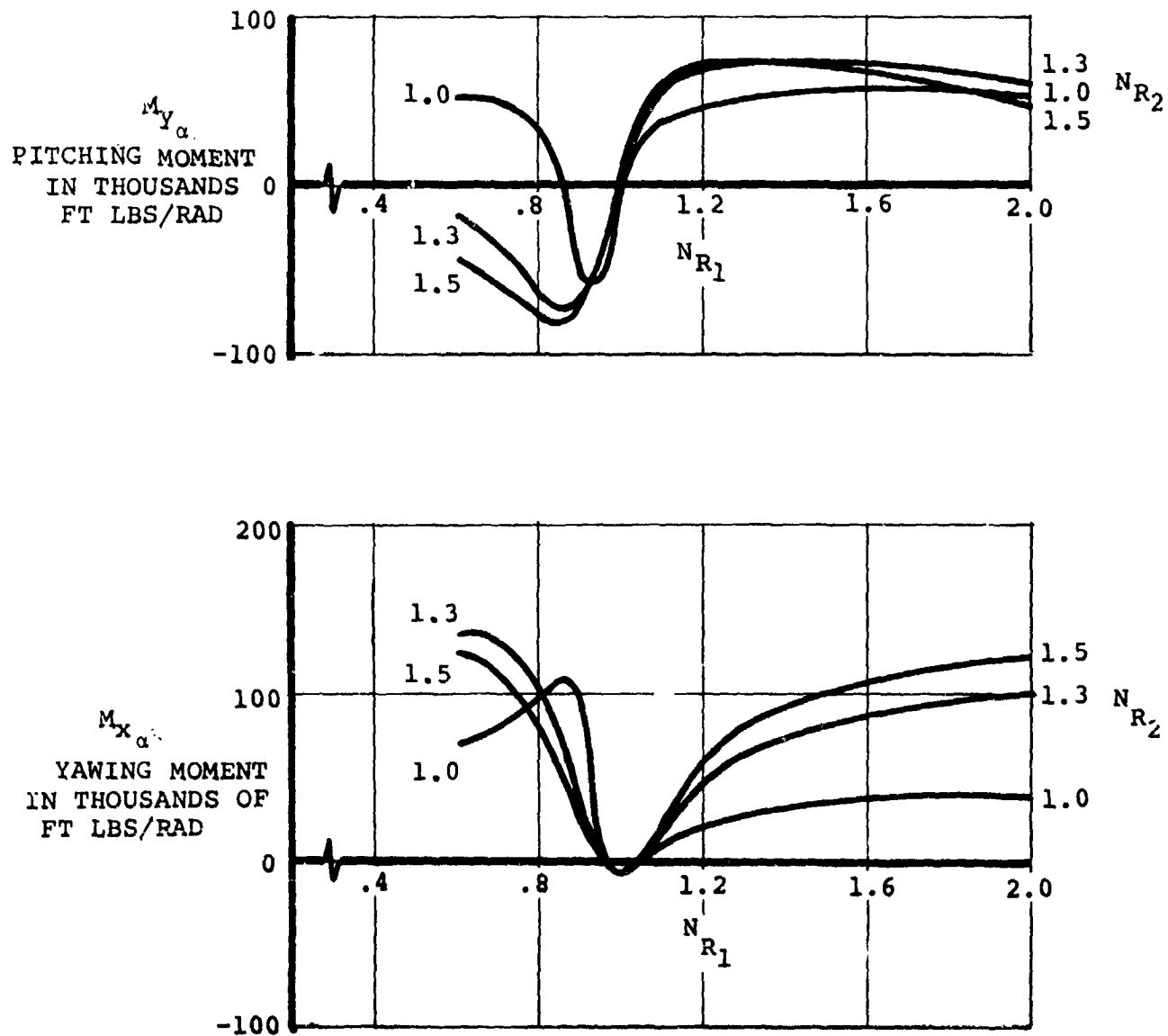
FIGURE 2.5.   VARIATION OF PROP/ROTOR PITCHING AND
YAWING MOMENT ALPHA DERIVATIVES WITH
RESPECT TO BLADE LEAD-LAG FREQUENCY.
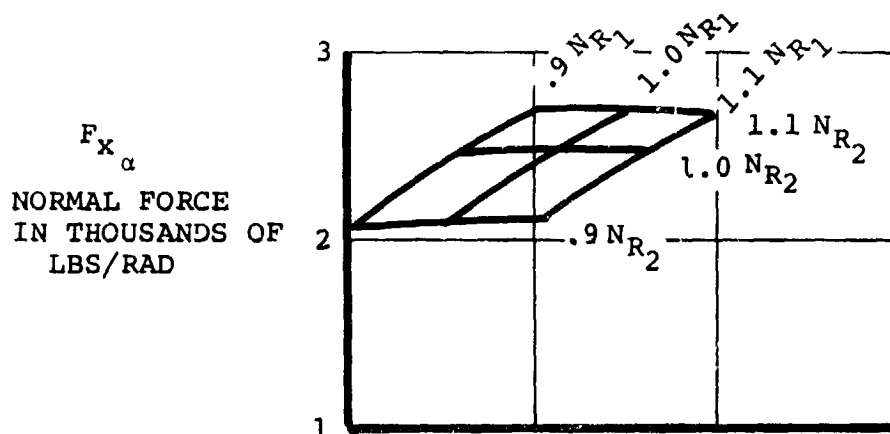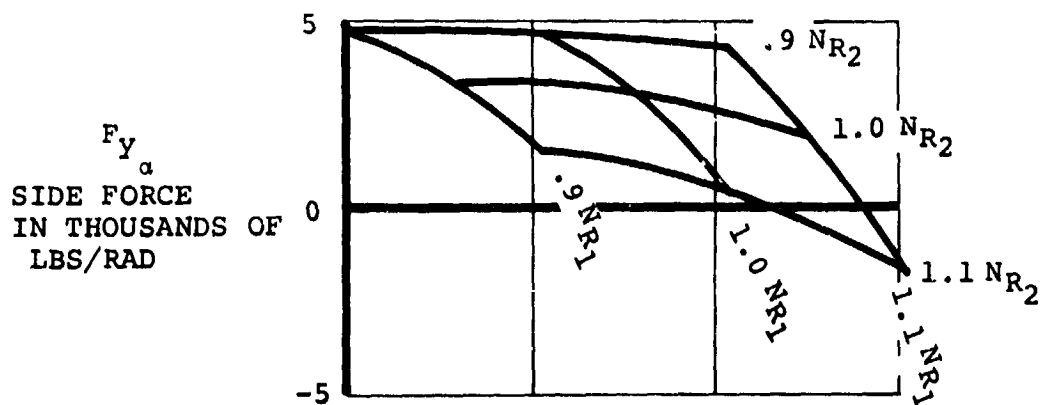
FIGURE 2.6 SENSITIVITY OF M222 26FT HINGELESS ROTOR ALPHA
DERIVATIVES FOR NORMAL AND SIDE FORCE TO
VARIATIONS IN BLADE LEAD-LAG AND FLAP NATURAL
FREQUENCIES
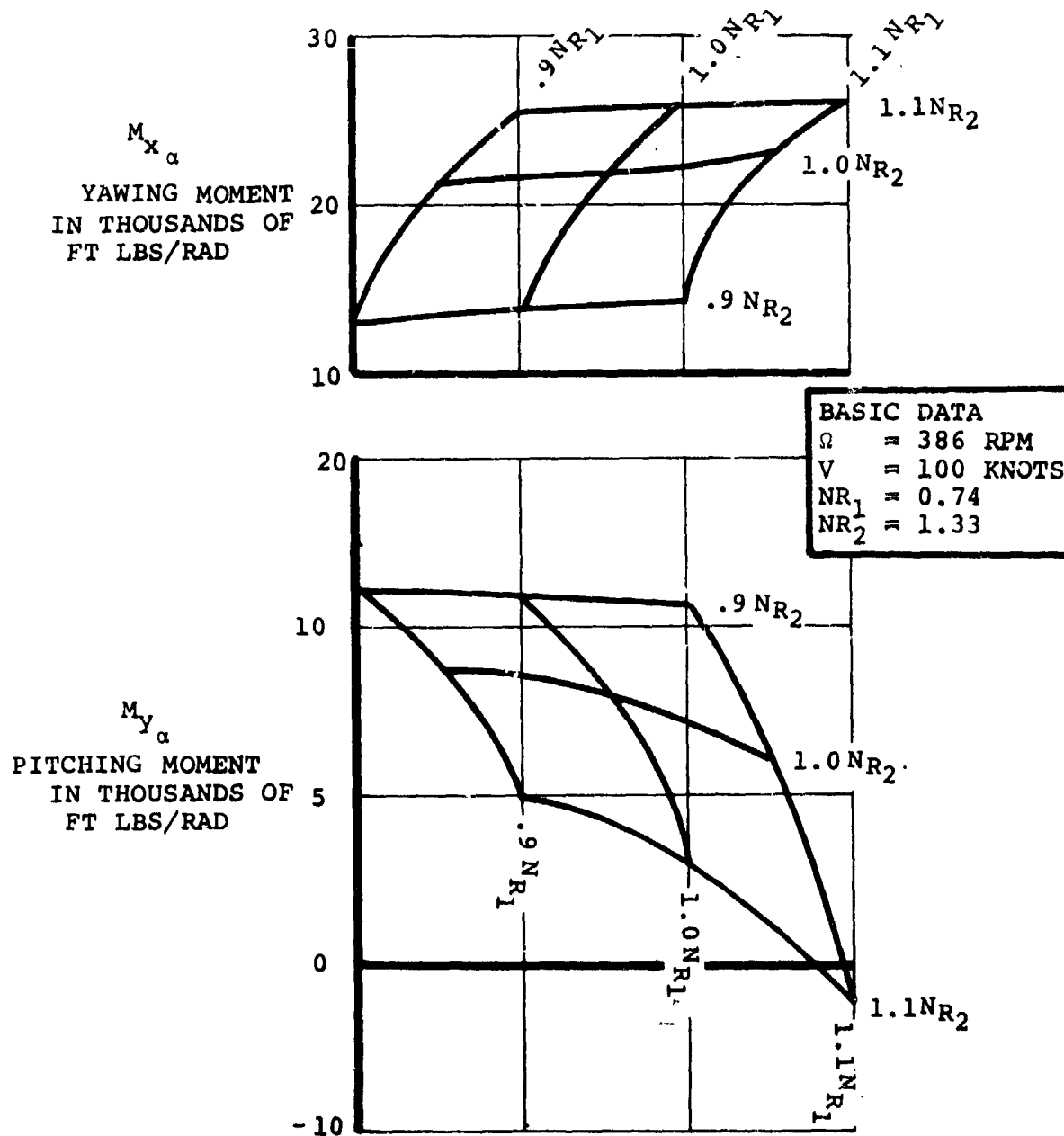
FIGURE 2.7 SENSITIVITY OF M222 26FT HINGELESS ROTOR ALPHA
DERIVATIVES FOR YAWING MOMENT AND PITCHING
MOMENT VARIATIONS IN BLADE LEAD-LAG AND FLAP
NATURAL FREQUENCIES

$F_{x_{A_1}}$
NORMAL FORCE
IN THOUSANDS OF
LBS/RAD

BASIC DATA
V = 100 KNOTS
Ω = 386 RPM
NR_1 = 0.74
NR_2 = 1.33

$F_{y_{A_1}}$
SIDE FORCE
IN THOUSANDS OF
LBS/RAD

FIGURE 2 8 SENSITIVITY OF M222 26FT DIAMETER HINGELESS ROTOR
CYCLIC DERIVATIVES FOR NORMAL AND SIDE FORCE TO
VARITIONS IN BLADE LEAD-LAG AND FLAP NATURAL
FREQUENCIES

15
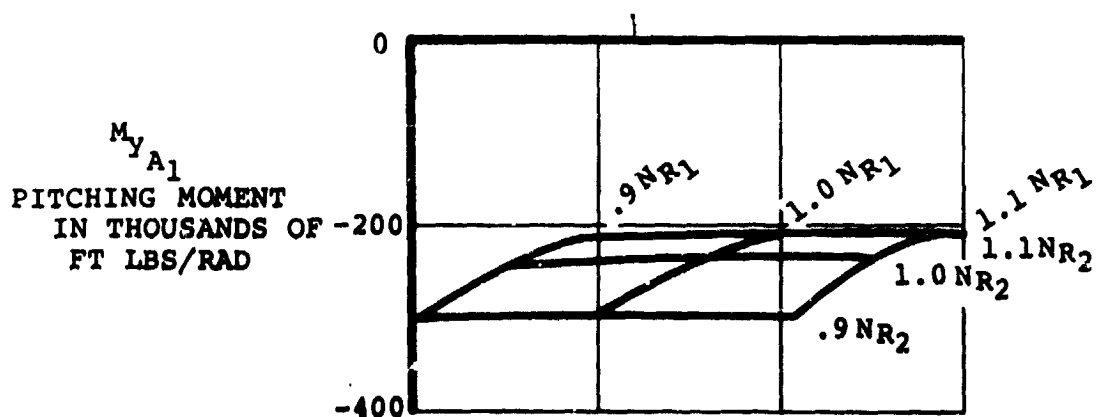
FIGURE 2.9 SENSITIVITY OF M222 26FT HINGELESS ROTOR CYCLIC
DERIVATIVES FOR YAWING AND PITCHING MOMENT, TO
VARIATIONS IN BLADE LEAD-LAG AND FLAP NATURAL
FREQUENCIES

significant, however, that when such a correspondence is observed there are differences in magnitude. In particular the side force due to A_1 is approximately four times larger than the normal force due to alpha. Figures 2.10 through 2.13 present the equivalent data for 200 knots and a similar pattern is noted.

These comparisons of alpha and cyclic derivatives are of considerable importance since they indicate that in some cases there are substantial differences which may be reflected as difficulties in selecting combinations of cyclic to zero out the hub forces and moments due to alpha. That is to say, even under the simplest static conditions there will be limits to the effectiveness of swashplate feedback load alleviation system.

2.4 EFFECTS OF ALTITUDE AND SPEED ON PROP/ROTOR DERIVATIVES

Figures 2.14 through 2.17 present carpets of alpha and A_1 derivatives for altitudes of 0, 10,000 and 20,000 feet and velocities of 100 through 300 knots. These plots display anticipated trends of increasing force and moment with velocity and reduction with altitude or density.

2.5 CORRELATION WITH TEST DATA

The derivative studies in the preceeding paragraphs were performed using the Boeing Vertol C-41 prop/rotor derivative program. This program has been used extensively in predicting and correlating with rotor derivatives and associated data obtained on test. In general prediction of trends is excellent

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F_{X_α}
NORMAL FORCE
IN THOUSANDS OF
LBS/RAD

16

.9N_{R2}

14

.9N_{R1}

1.0N_{R1}

1.0N_{R2}

1.1N_{R2}

12

1.1N_{R1}

BASIC DATA
Ω = 386 RPM
V = 200 KNOTS
N_{R_1} = 0.76
N_{R_2} = 1.36

4

.9N_{R1}

1.0N_{R1}

1.1N_{R1}

0

.9N_{R2}

F_{Y_α}
SIDE FORCE IN
THOUSANDS OF
LBS/RAD

-4

1.0N_{R2}

1.1N_{R2}

-8
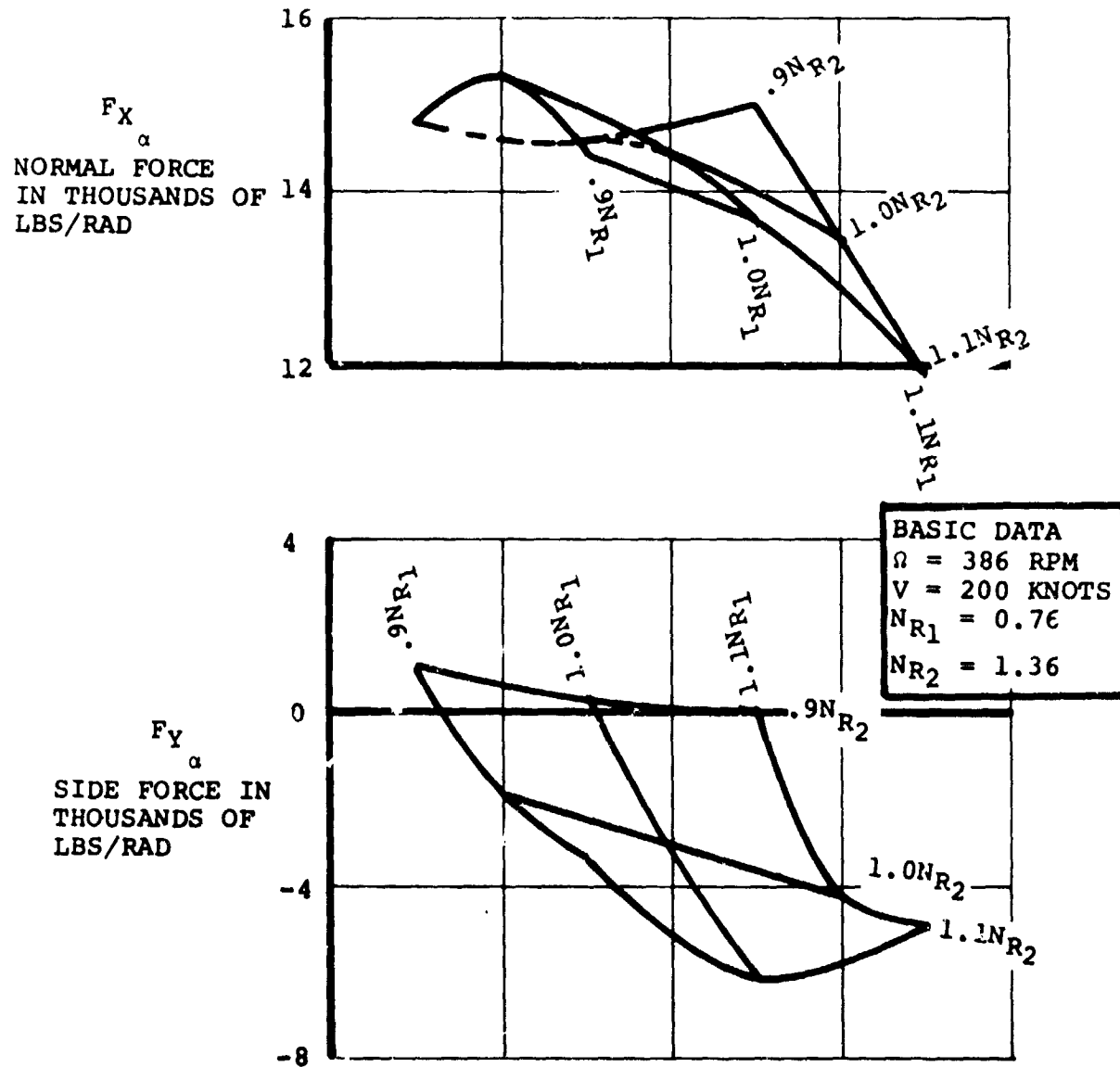
FIGURE 2.10.   SENSITIVITY OF M222 26-FT DIAMETER HINGE-
LESS ROTOR NORMAL AND SIDE FORCE ALPHA
DERIVATIVES TO BLADE LEAD-LAG AND FLAP
NATURAL FREQUENCIES.

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BASIC DATA
Ω = 386 RPM
V = 200 KNOTS
N_{R_1} = 0.76
N_{R_2} = 1.36

M_{X_α}

YAWING MOMENT
IN THOUSANDS OF
FT LBS/RAD

M_{Y_α}

PITCHING MOMENT
IN THOUSANDS OF
FT LBS/RAD
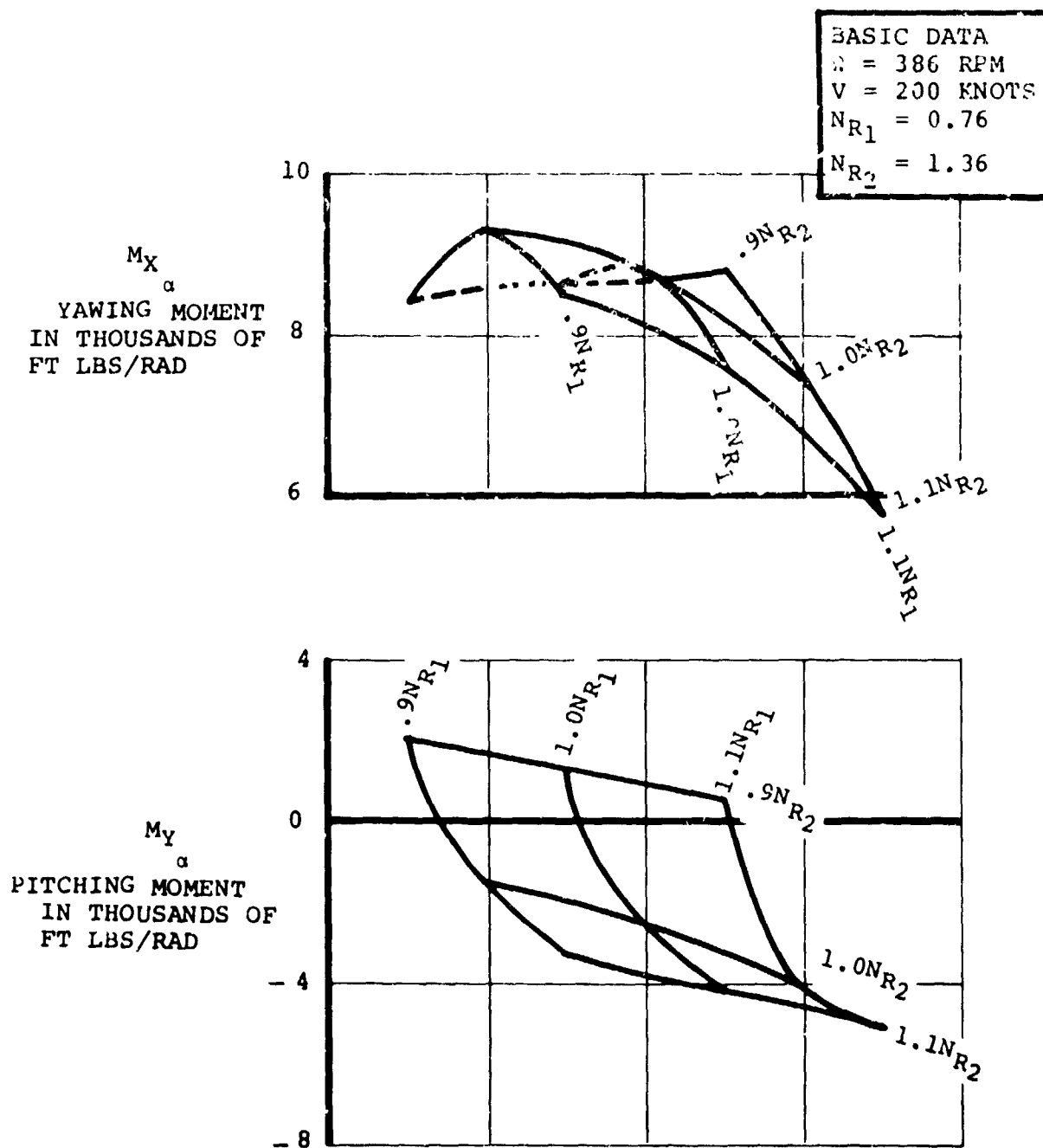
FIGURE 2.11. SENSITIVITY OF M222 26-FT DIAMETER HINGE-
LESS ROTOR YAWING AND PITCHING MOMENT ALPHA
DERIVATIVES WITH RESPECT TO BLADE LEAD-LAG
AND FLAP NATURAL FREQUENCIES.

19
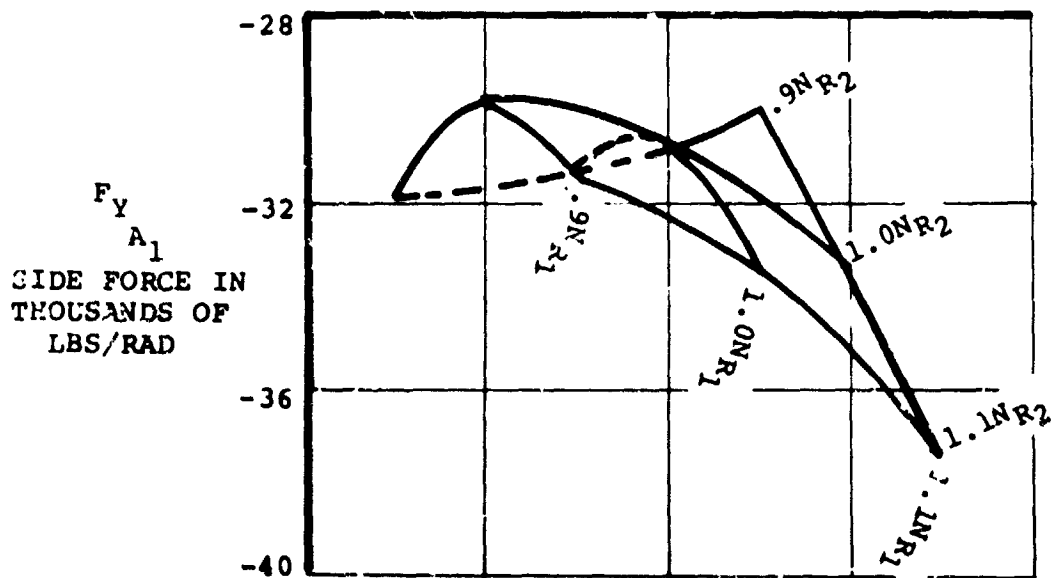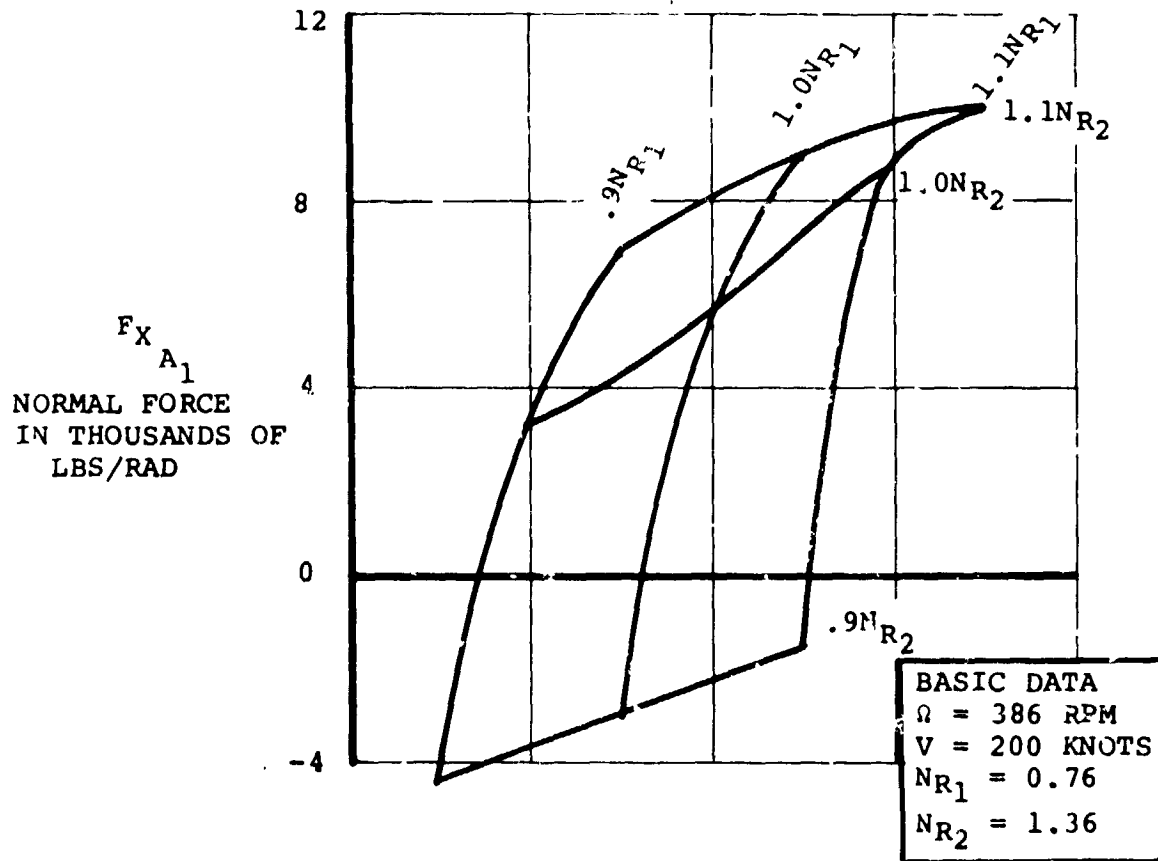
FIGURE 2.12. SENSITIVITY OF M222 26-FT DIAMETER HINGE-LESS ROTOR NORMAL AND SIDE FORCE CYCLIC DERIVATIVES WITH RESPECT TO BLADE LEAD-LAG AND FLAP NATURAL FREQUENCIES.

$$M_X \atop A_1$$

YAWING MOMENT
IN THOUSANDS OF
FT LBS/RAD

BASIC DATA
Ω = 386 RPM
V = 200 KNOTS
N_{R_1} = 0.76
N_{R_2} = 1.36

$$M_Y \atop A_1$$

PITCHING MOMENT
IN THOUSANDS OF
FT LBS/RAD
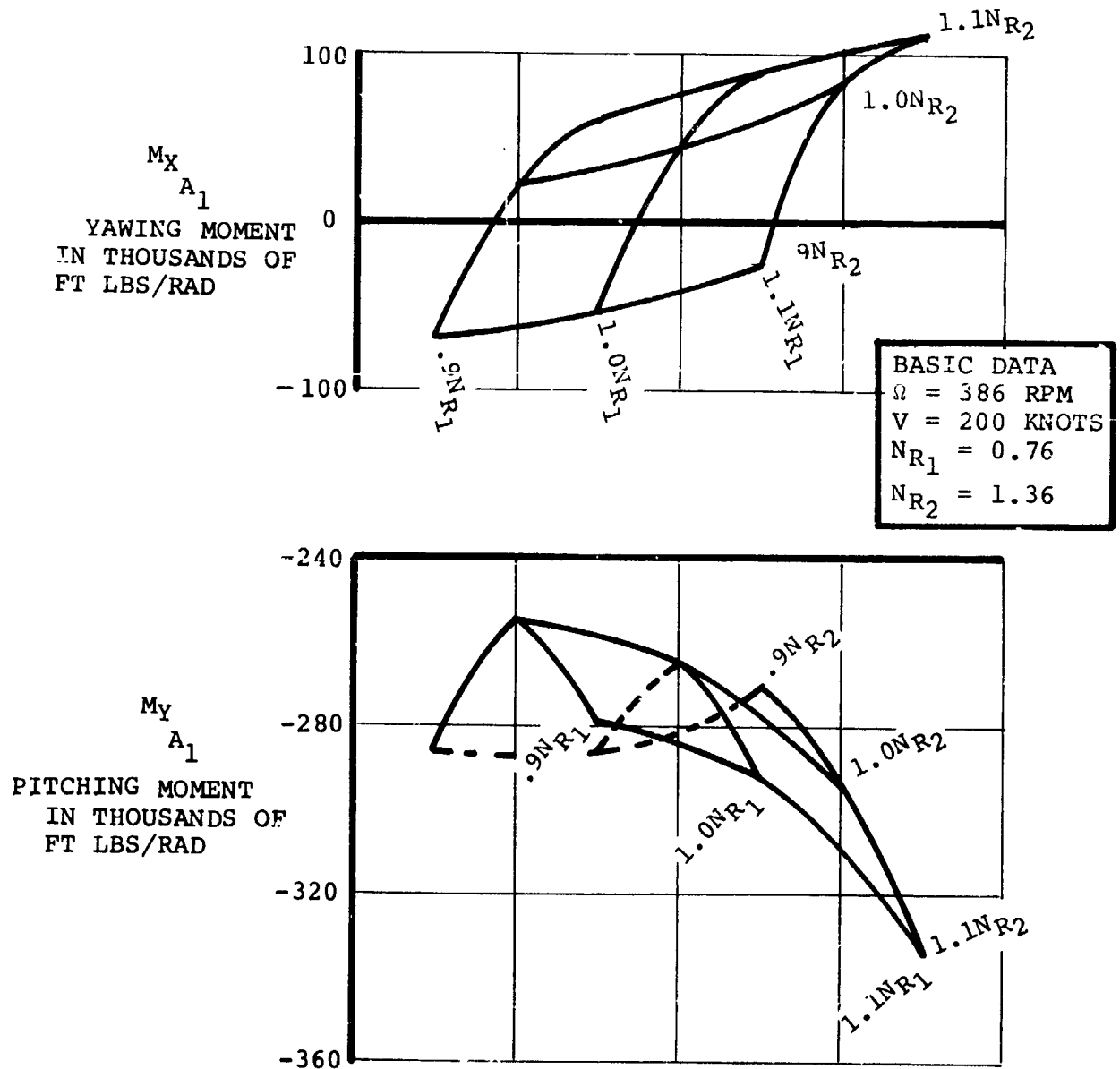
FIGURE 2.13.   SENSITIVITY OF M222 26-FT DIAMETER HINGE-
LESS ROTOR YAWING AND PITCHING MOMENT CYCLIC
DERIVATIVES WITH RESPECT TO BLADE LEAD-LAG
AND FLAP NATURAL FREQUENCIES.

21

D222-10060-3
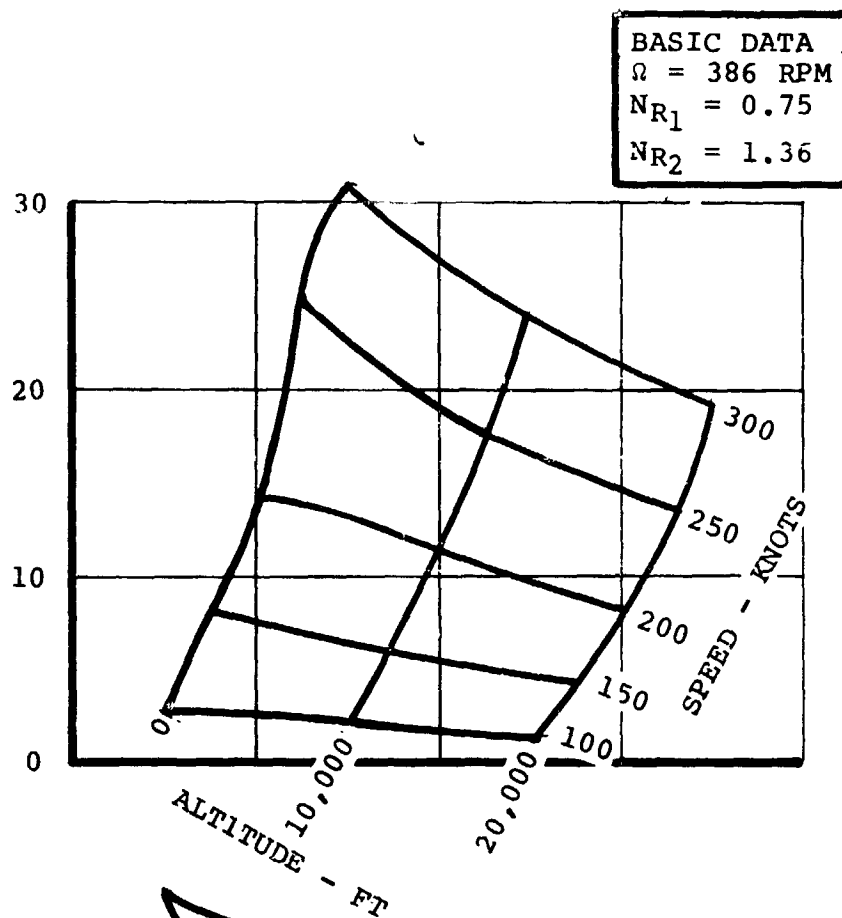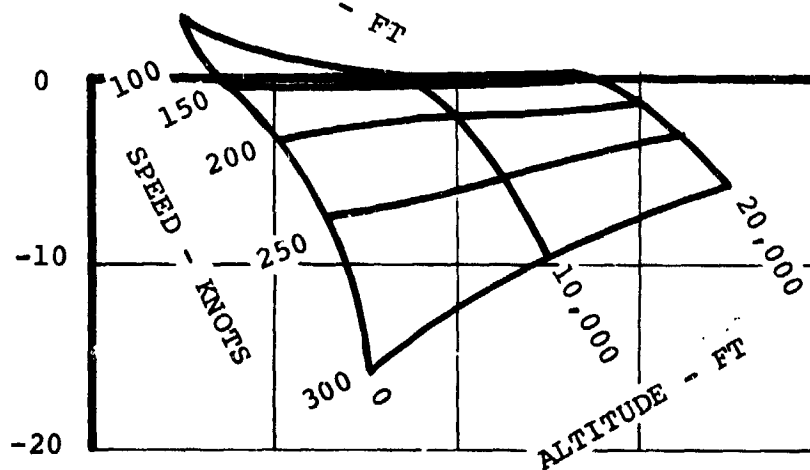
FIGURE 2.14.   SENSITIVITY OF M222 26-FT DIAMETER HINGE-
LESS ROTOR NORMAL AND SIDE FORCE ALPHA
DERIVATIVES WITH RESPECT TO ALTITUDE AND
VELOCITY.

22
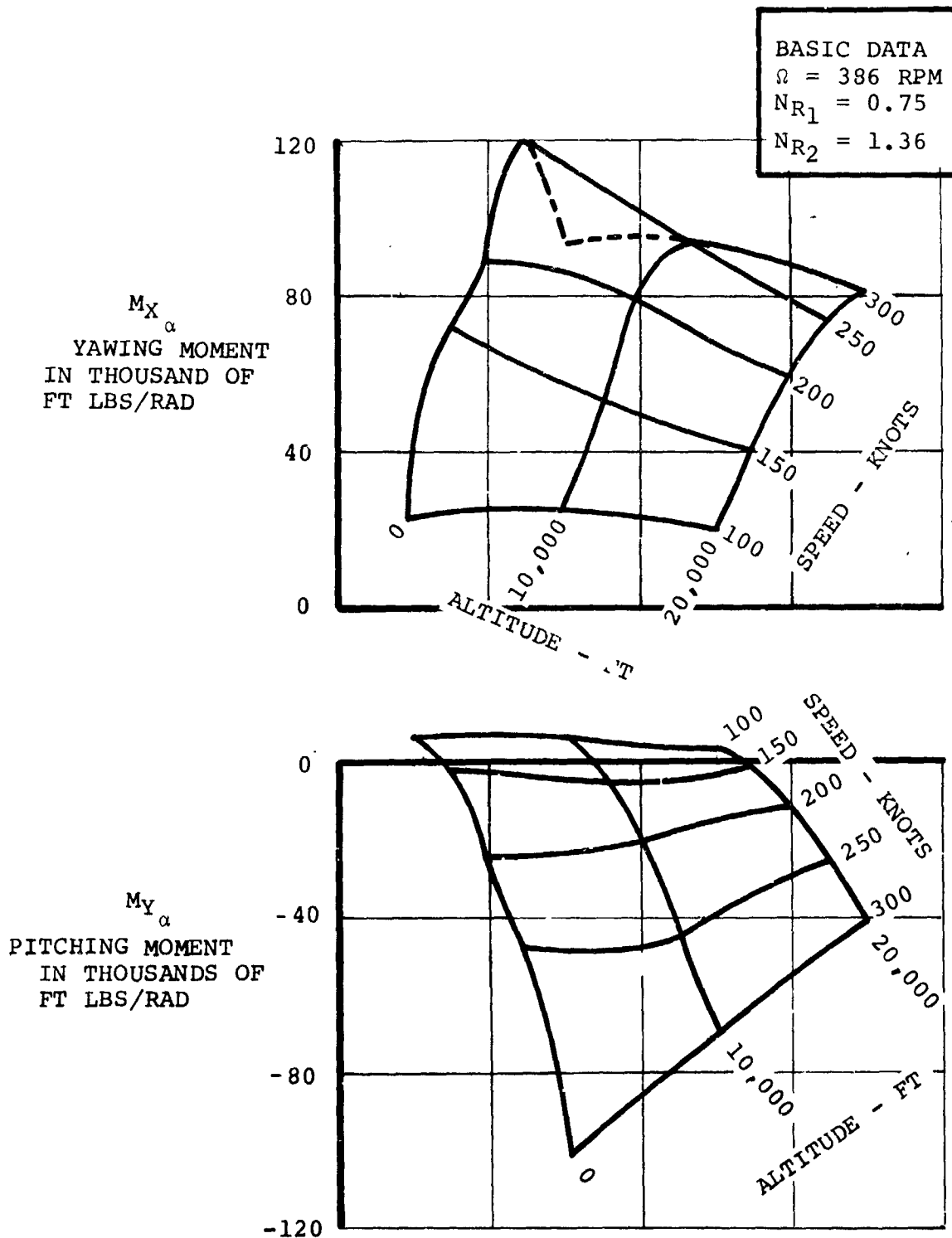
FIGURE 2.15. SENSITIVITY OF M222 26-FT DIAMETER HINGE-LESS ROTOR YAWING AND PITCHING MOMENT ALPHA DERIVATIVES WITH RESPECT TO ALTITUDE AND VELOCITY.
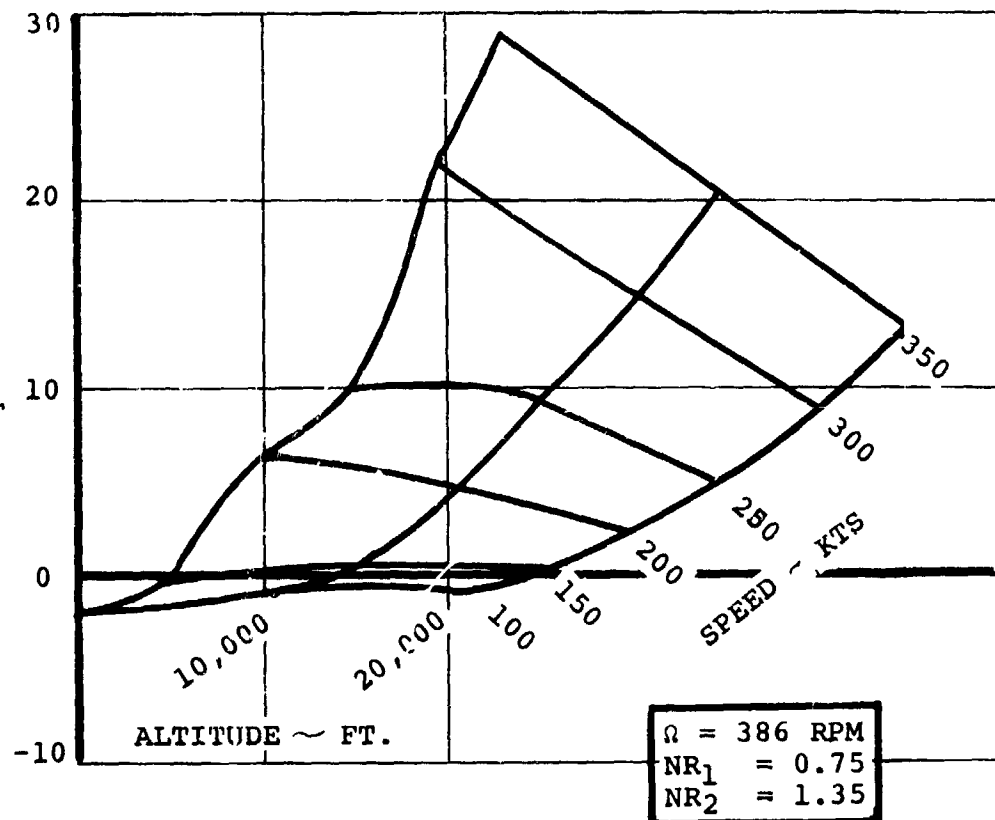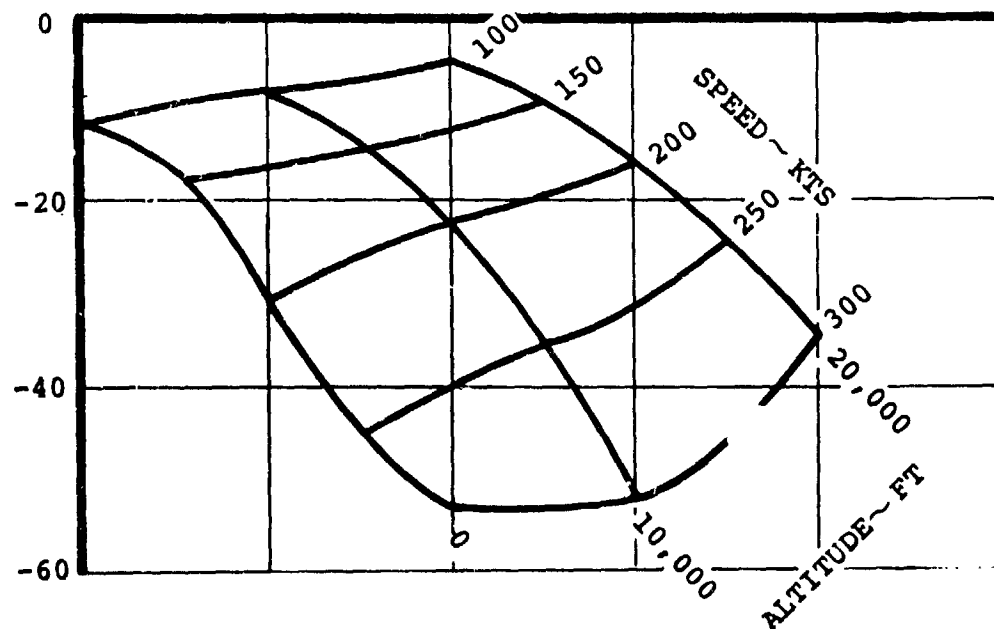
FIGURE 2.16 SENSITIVITY OF M222 26FT DIAMETER ROTOR
NORMAL FORCE AND SIDE FORCE CYCLIC
DERIVATIVES TO ALTITUDE AND AIRSPEED

$M_{X_{A_1}}$
YAWING MOMENT
IN THOUSANDS OF
FT LBS/RAD

$M_{Y_{A_1}}$
PITCHING MOMENT
IN THOUSANDS OF
FT LBS/RAD

$\Omega = 386$ RPM
$NR_1 = 0.75$
$NR_2 = 1.35$
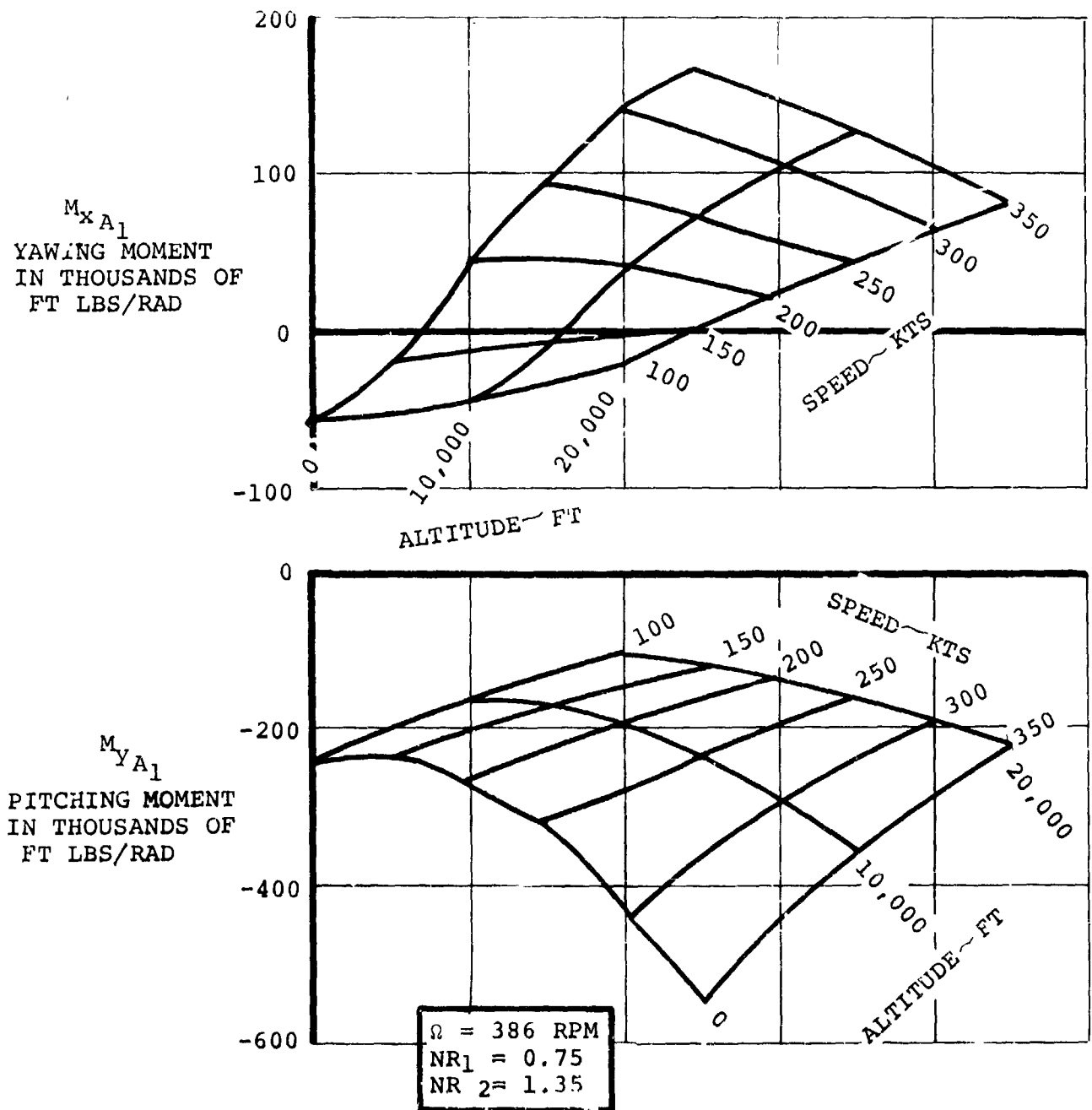
FIGURE 2.17. SENSITIVITY OF M222 26FT DIAMETER ROTOR YAWING
AND PITCHING MOMENT CYCLIC DERIVATIVES TO
ALTITUDE AND AIRSPEED

25

D222-10060-3

with quite good agreement in absolute magnitudes.

2.5.1   Model 213 Four Blade Hingeless Rotor Correlation

Figure 2.18 presents correlation with rotor derivatives measured on a 1/9 scale dynamically similar model of a tilt/stowed rotor conversion model. In this test the rotor hub forces and moments were carefully measured over a range of RPM in which the lead-lag modal frequency progressed from less than 1 per rev at 900 RPM to values significantly greater than 1 per rev as the rotor was feathered. The measured values confirm the predicted behavior trend and the quantitative correlation is also excellent.

2.5.2   Correlation With Model 222 26-Foot Diameter Rotor Test
         in NASA-Ames 40 X 80-Foot Tunnel

Figure 2.19 shows the schematic of the windmilling test stand and its instrumentation. Test data were obtained from strain gages mounted on the outer portion of the wing as shown, and calibrated to measure normal force, pitching moment and yawing moment. Comparison with test data was made by calculating the moments about the wing strain gage locations using forces and moments predicted by the C-41 program. The results of this comparison for alpha derivatives are given in Figure 2.20 and for cyclic pitch derivatives in Figures 2.21 and 2.22. The analysis did not attempt to account for force and moment contributions from nacelle and wing aerodynamic interference. Nevertheless, quite good correlation is observed. These plots also show the values of derivatives

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Test Points ●

Theoretical Predictions

― ― ― ― Rigid Blades
―・― Flap Only
―――― Flap & Lead Lag

$$\frac{dC_L}{d\alpha} = 5.7$$

Definitions

Normal Force $= 1/2\rho\pi R^2 V^2 C_{N\alpha}\cdot\alpha$
Side Force $= 1/2\rho\pi R^2 V^2 C_{Y\alpha}\cdot\alpha$
Pitching Moment $= \rho\pi R^3 V^2 C_{m\alpha}\cdot\alpha$
Yawing Moment $= \rho\pi R^3 V^2 C_{n\alpha}\cdot\alpha$
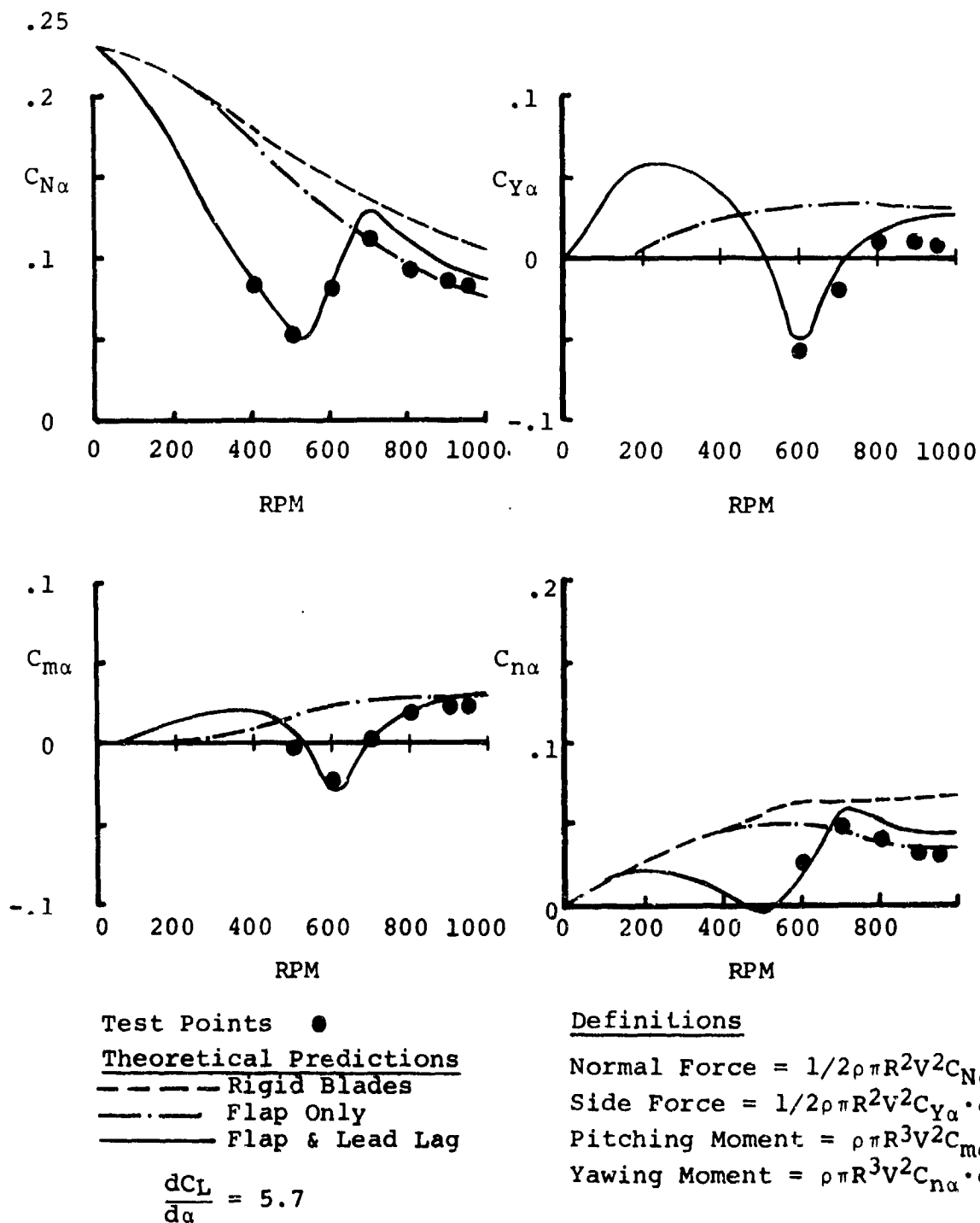
Figure 2.18. Model 213 1/9-Scale Conversion Model - 85 Ft/Sec
Derivative Variation with RPM

27
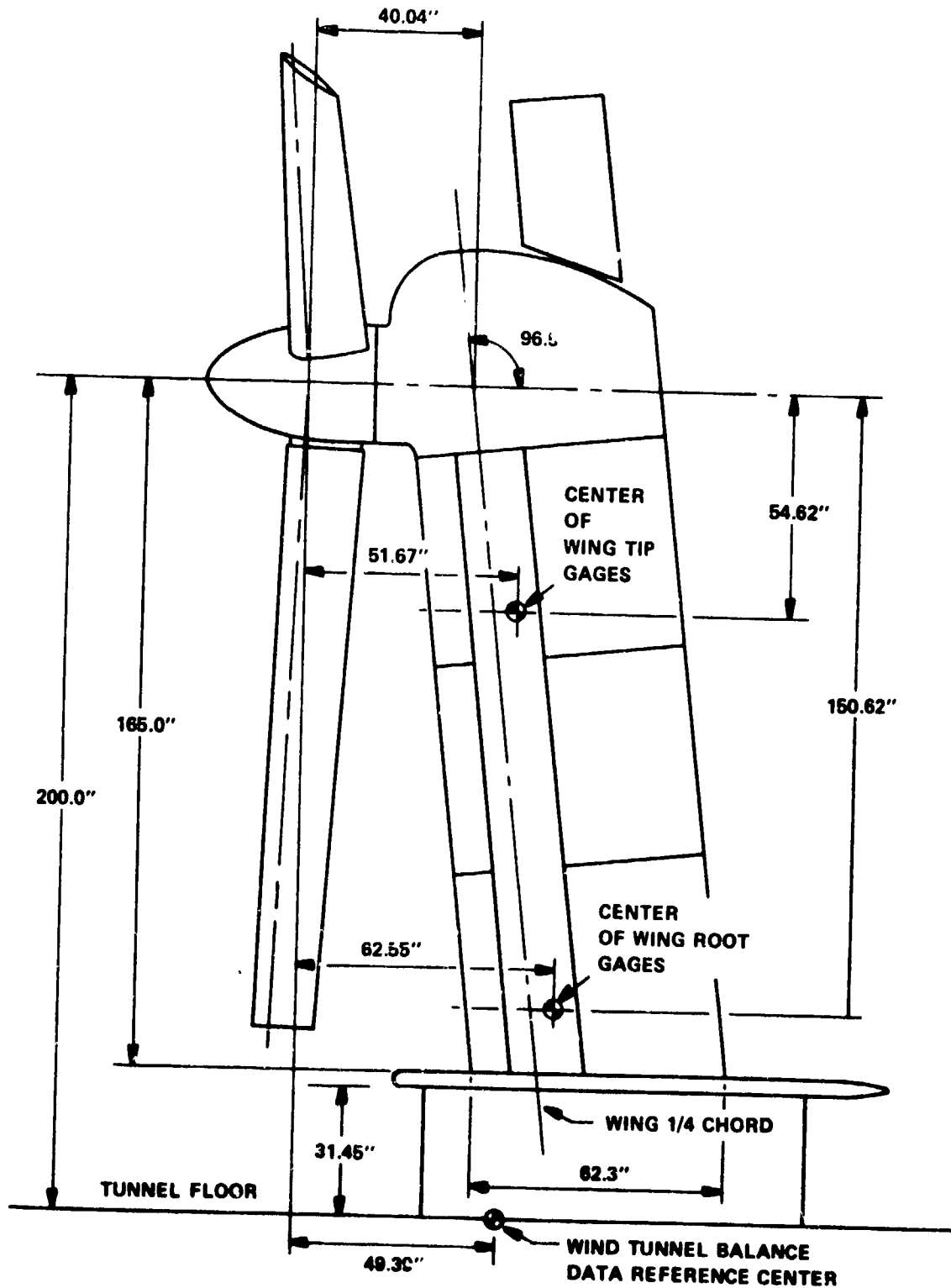
FIGURE 2.19.   26 FT. ROTOR TEST STAND IN NASA'S 40' X 80' TUNNEL
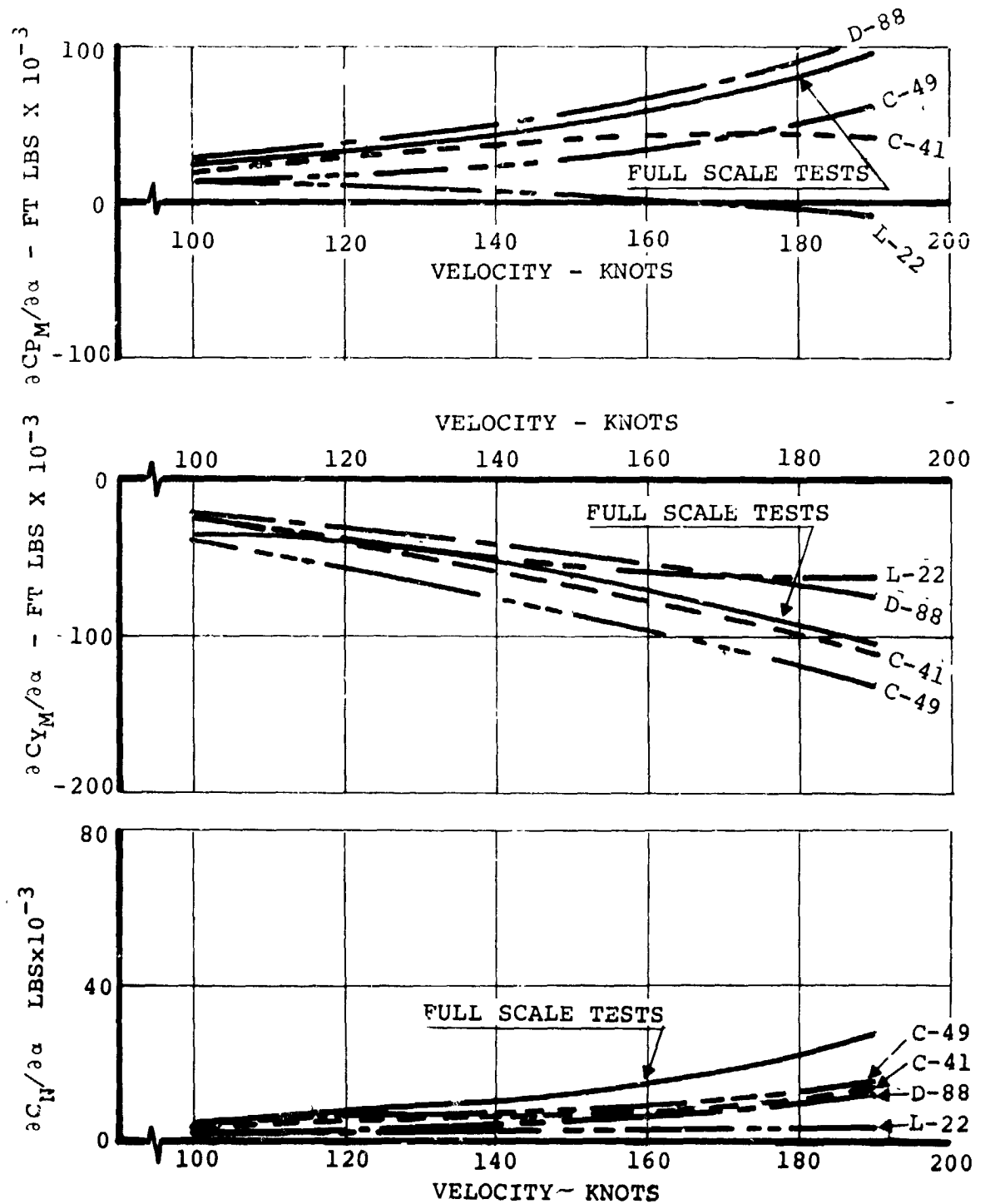
FIGURE 2.20.   CORRELATION OF 26 FT. ROTOR TEST DATA WITH
VARIOUS ROTOR DERIVATIVE PROGRAMS –
α DERIVATIVES

29
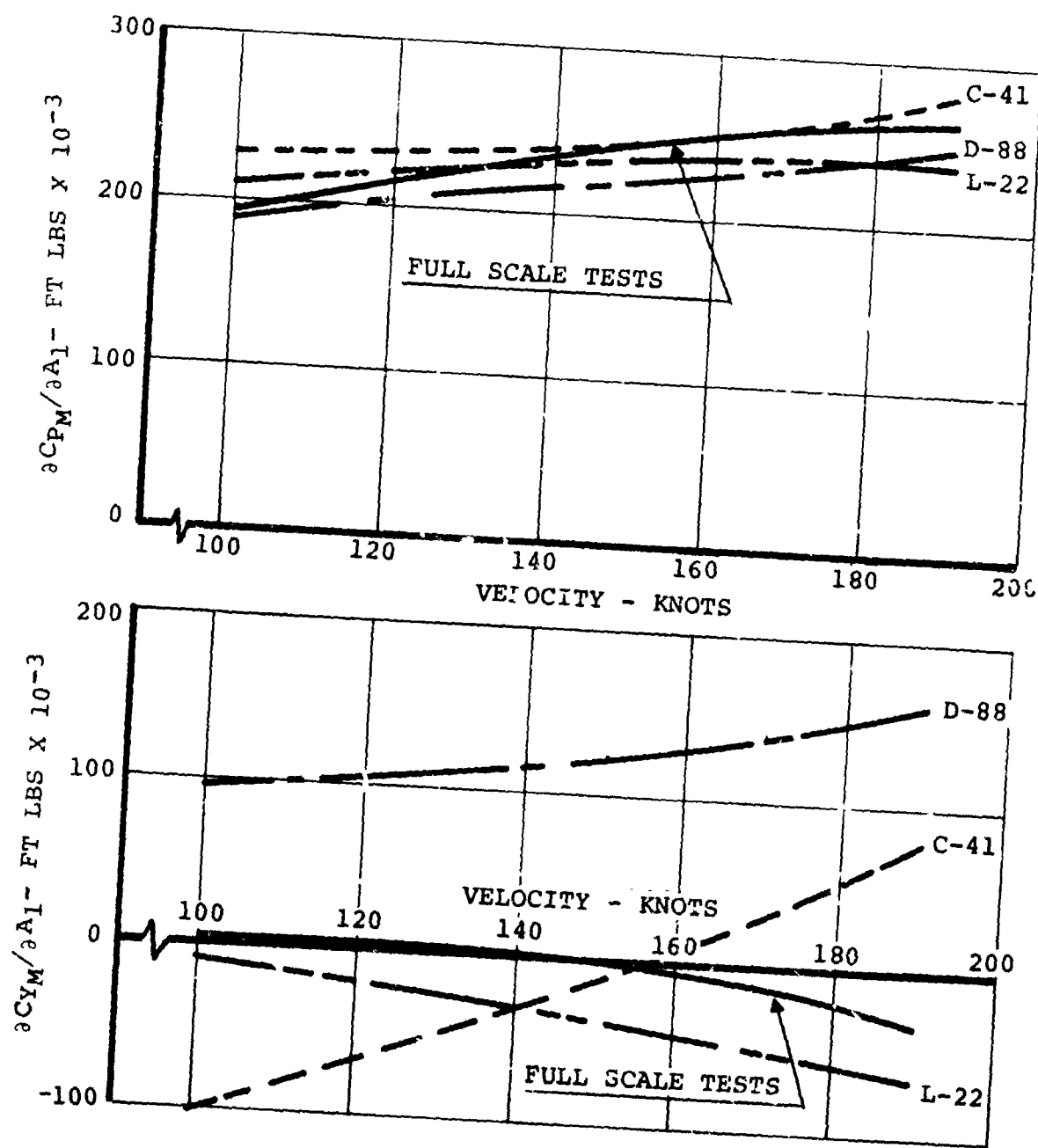
FIGURE 2.21. CORRELATION OF 26 FT. ROTOR TEST DATA WITH VARIOUS ROTOR DERIVATIVE PROGRAMS - CYCLIC MOMENT DERIVATIVES
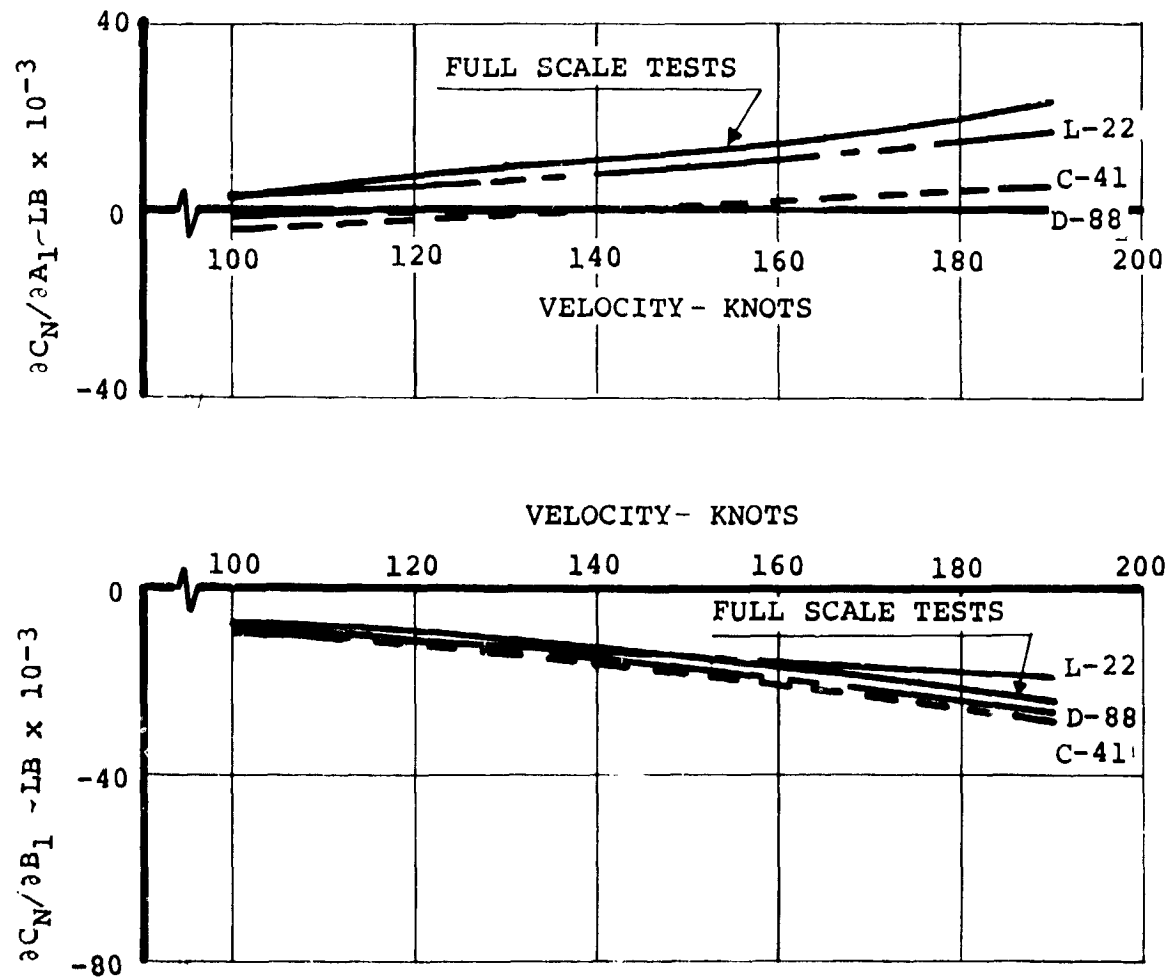
FIGURE 2.22. CORRELATION OF 26 FT. ROTOR TEST DATA WITH
VARIOUS ROTOR DERIVATIVE PROGRAMS - CYCLIC
FORCE DERIVATIVES

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predicted by several other programs. These include D-88 which
is a blade loads program which accounts for compressible non-
linear downwash and L-22 which uses linear airfoil theory and
uncoupled flap-lag freedoms. C-49 accounts for unsteady aero-
dynamics while C-41 uses a linear representation. C-41 and C-49
use a modal represen_ation of blade freedoms (2 coupled flap-lag
modes) while D-88 and L-22 make use of a finite element discrete
mass representation.

The rotor derivative data was also compared with C-41 using a
total unresolved moment approach. Total moments about the center
of the wing tip gages and the reference azimuth position (orien-
tation of the moment vector in the rotor disc plane) were cal-
culated from the C-41 hub forces and moments and compared with
test results (Figure 2.23). The interesting conclusion
which is not apparent from the resolved forces and moments is
that the total moment is predicted well but there are slight
differences in the reference azimuth position.

2.5.3 Correlation with Model 222 1/4.622 Scale Model Data

The subject model is a dynamically similar version of the M222.
The test data presented in Figures 2.24 and 2.25 were taken
with the model mounted on a pedestal in the tunnel. The rotors
were given angles of attack to the free stream by pitching the
complete model with zero sideslip angle and yawing the model
at zero angle of attack. The yawing data contains minimal
wing induced flow effects and comparison with the pitch data

32
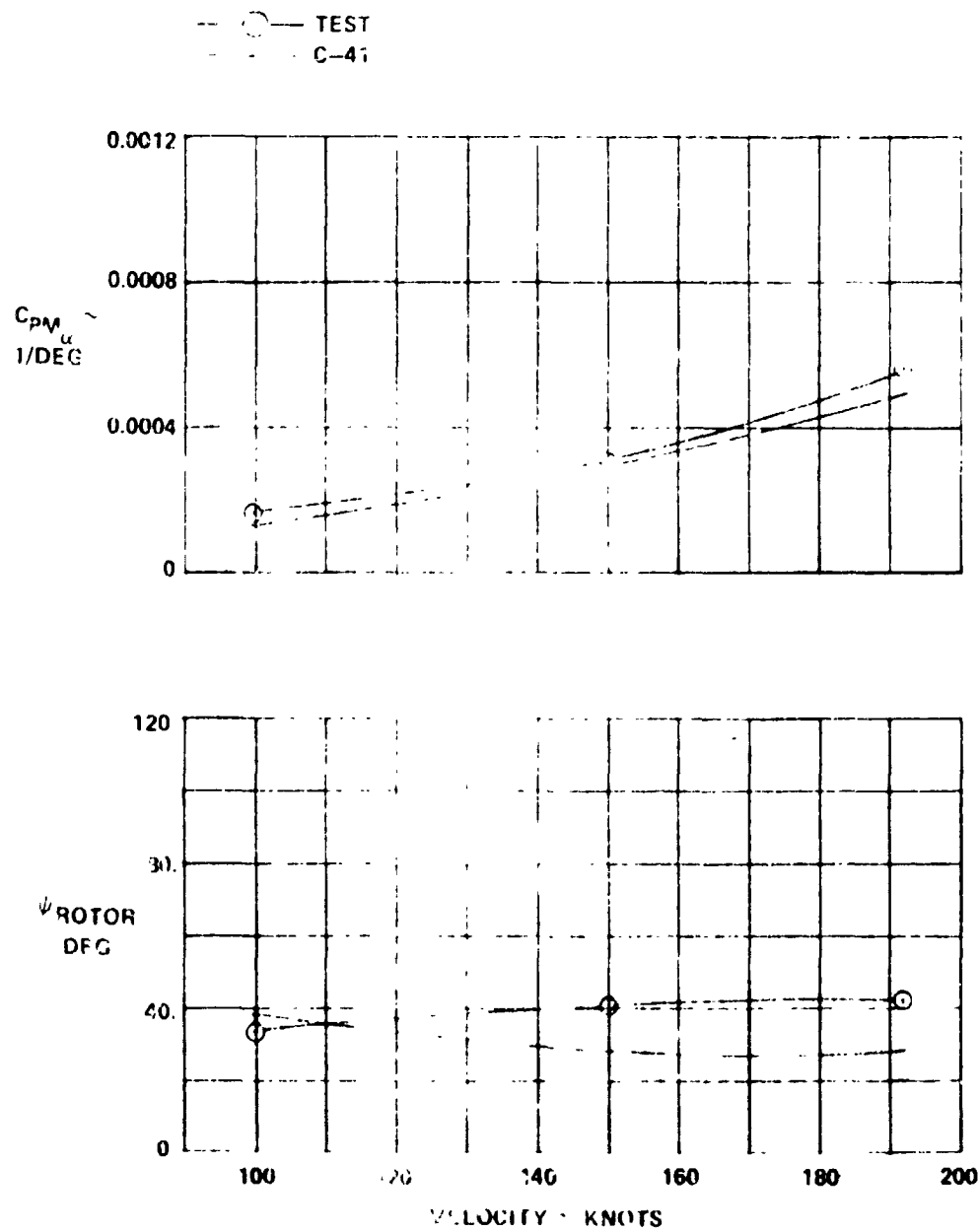
FIGURE 2.23. ROTOR MOMENT AND AZIMUTH ANGLE DUE TO ANGLE OF ATTACK - CORRELATION WITH 26-FT ROTOR DATA (ABSTRACTED FROM M222 PROPOSAL-BOEING D222-10050-2)
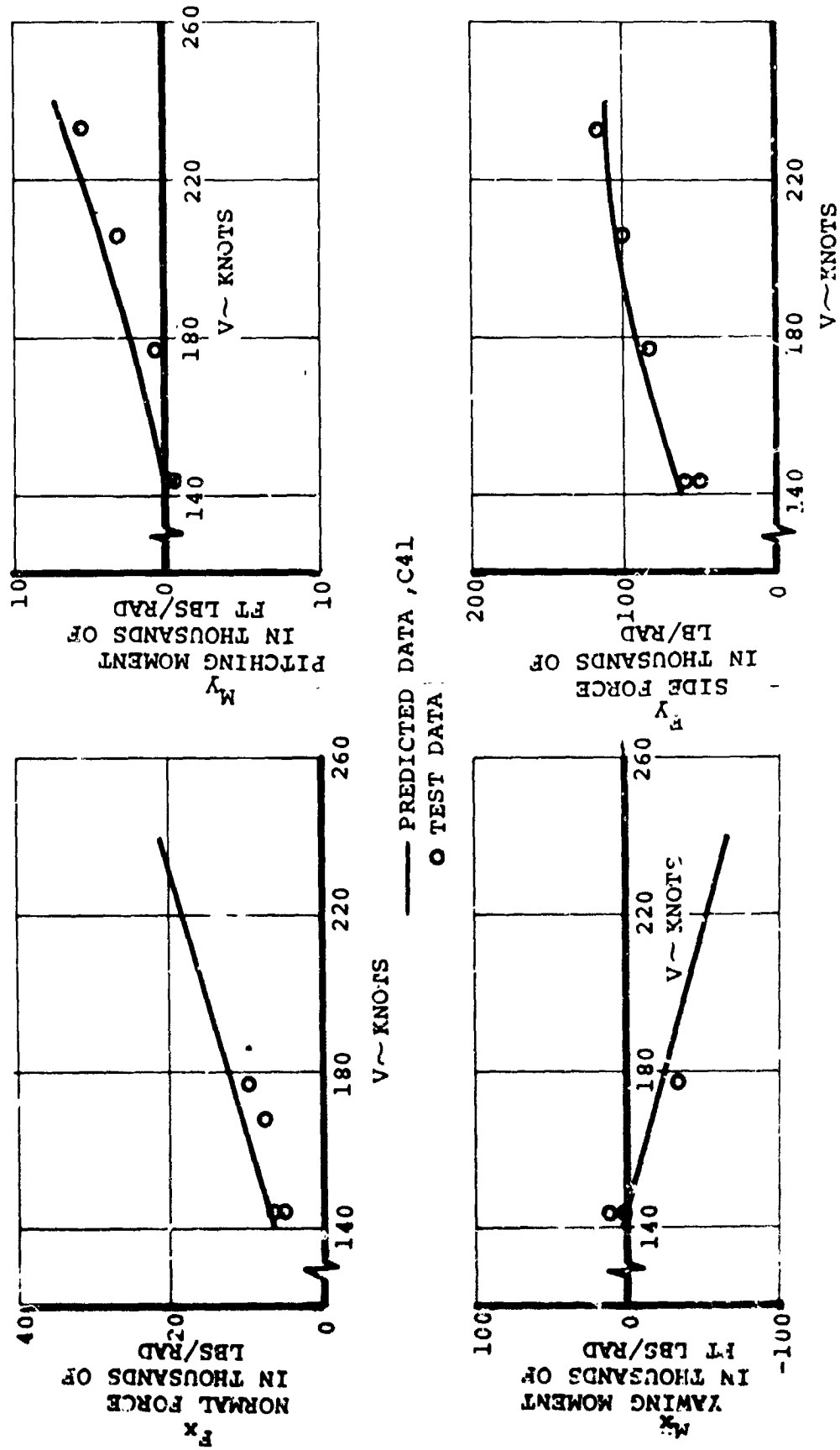
FIGURE 2.24. COMPARISON OF CALCULATED AND TEST ROTOR HUB FORCE AND MOMENT DERIVATIVES FOR M-222 1/4.622 SCALE MODEL (YAW SWEEP) 386 RPM

indicates the importance of induced flow on the rotor forces and moments. Forces and moments were computed for the isolated rotor and it is seen from Figure 2.24 that correlation with test data is excellent when wing induced effects are small; in Figure 2.25 wing effects introduce perceptible shifts which increase with dynamic pressure.

2.6   CONCLUSIONS FROM DERIVATIVE STUDY

It is seen that there is a strong similarity between the derivatives for angle of attack and cyclic; however, the differences are such that it is probably impossible using cyclic to completely null out simultaneously all hub forces and moments due to angle of attack. The variation of cyclic derivatives as a function of lag frequency is similar to that for angle of attack. Comparison of analytical data from isolated rotor with test shows that correlation is good when isolated rotor assumptions are effectively, met, but that induced flow effects are important when this is not the case.
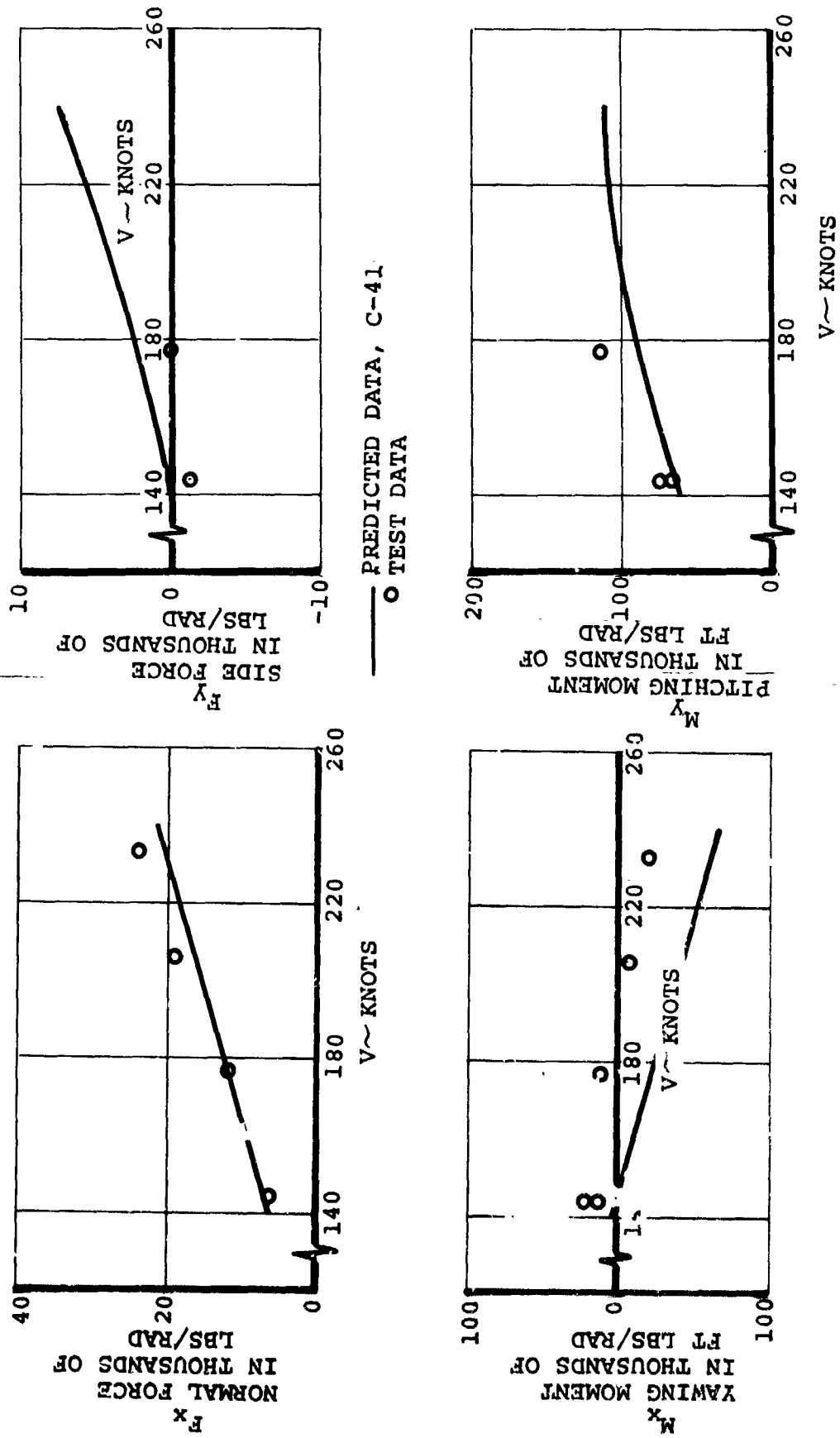
FIGURE ?.2.5. COMPARISON OF CALCULATED AND TEST ROTOR HUB FORCE AND MOMENT DERIVATIVES FOR M-222 1/4.622 SCALE MODEL (PITCH SWEEP) $\Omega = 386$ RPM

3.   BLADE LOAD ALLEVIATION AND STABILITY AUGMENTATION SYSTEMS

3.1   BACKGROUND AND OBJECTIVES OF STUDY

Tilting prop/rotor type aircraft experience significant blade
loads as a result of non-axial flow in transition from hover
to the cruise configuration, and in transient conditions such
as maneuvers, gusts, sideslip, etc. However, since cyclic
pitch is a basic feature of most tilt rotor control systems,
it provides a means to significantly reduce the severity of
loading conditions associated with skewed flow. This is
accomplished in two ways. The first is to schedule the appli-
cation of controlled amounts of longitudinal and lateral
cyclic as a function of flight condition. The second, which
is the primary topic of this report, is the automatic applica-
tion of cyclic to reduce loads in amounts proportional to the
deviations from the scheduled flight program, or to some
equivalent loading in the structure caused by the deviation.
Such a system will not only reduce blade loads, but will at the
same time reduce the associated hub force and moment deriva-
tives, thus increasing the static stability margin of the air-
craft.

The objective of this study is to explore the use of load
alleviation systems in a typical tilt rotor design, taking

into account those factors which might adversely affect performance in a practical situation. These include hardware characteristics, sensors and actuators, and the impact of dynamic transient effects as well as idealized steady state alleviation. System authority is also discussed for its impact on effectiveness at different flight conditions. The ability of a feedback control system working through the swashplate to influence the following will be analyzed:

O   Reduction of blade loads and hub forces and moments under steady maneuvers and gust encounters

O   Improvement of flying qualities by reducing destabilizing forces and moments from the rotors; improvement of short period response and pilot workload

O   Alleviation of airframe structural loads

O   Improve ride qualities by reduction of gust response accelerations

3.2   TECHNICAL BASIS FOR USE OF CYCLIC PITCH FEEDBACK IN LOAD ALLEVIATION

The predominant cause of vibratory loading in prop/rotors is blade dynamic response to cyclical angle of attack changes associated with non-axial flow caused by shaft tilt to the free stream or with cyclic pitch of the blade due to tilt of the swashplate. That is to say, in a propeller or rotor

whose shaft is inclined at an angle α to the free stream each blade experiences a sinusoidal increment of incidence of amount $\alpha \sin \Omega t$. It also experiences a sinusoidal variation in relative velocity over the blade, and both these effects combine to give a variation in dynamic pressure and in angle of attack. The net effect is to produce cyclical perturbation in the blade loads and blade dynamic response. Associated with blade response are corresponding shears, bending moments and strains. Cyclic pitch imposes a 1 per rev variation in incidence and has accordingly much the same effect as shaft incidence except that the angle of attack increment is uniform across the blade and there is no directly associated variation in blade dynamic pressure. Cyclic pitch in appropriate amounts is, therefore, used to trim out the angle of attack variations caused by shaft tilt to the relative wind. The use of cyclic pitch to trim out blade loads and for stability augmentation is established practice in the helicopter field, and the extension to tilt rotor applications is clearly indicated. There is, however, minimal discussion of such topics as scheduling of cyclic to minimize blade loads for normal flight conditions. The emphasis is on the use of automatic feedback cyclic control to counteract loa' occurring due to off-schedule conditions.

Such conditions occur during maneuvers and turbulence when
the rotor experiences temporary departures from the trimmed
unaccelerated flight condition.

3.3   TEST DEMONSTRATION OF SWASHPLATE FEEDBACK SYSTEMS

Two test programs were conducted in 1972 in which the use of
swashplate feedback for load alleviation was demonstrated.
The first,in May,was performed using a 1/9.622 scale model of
the Model 222 rotor mounted on NASA wing in the Princeton
Tunnel. The sensor system used consisted of strain gages
measuring pitching moment and yawing moment in the wing. The
system was demonstrated for static situations (i.e., steady
wing angle of attack) and also for simulated long period gust
conditions using the gust generating capability of the
Princeton Tunnel. The results of this test indicated that
substantial reductions in blade response were available with
the correct selection of azimuth and gain. The results of
this test are reported in Boeing Document D222-10047-1
(Reference 3.1). In September of the same year, the full
scale version of the above model was tested in the NASA Ames
40 X 80-foot tunnel with a similar feedback system operative.
This test also showed that substantial reductions in blade
loads could be achieved using a swashplate feedback system.
The results of this test are given in Boeing Document D222-
10059-1 dated March 1973, (Reference 3.2). The results of
both these tests tend to confirm the results presented in
this report.

3.4 CANDIDATE SYSTEMS: CHOICE OF SENSORS

The principal feature differentiating one load alleviation system from another is the signal sensed and fed back through the swashplate. A number of potentially viable signals and sensors are tabulated in Table 3.1 along with the advantages and disadvantages of each system.

Of the sensors listed, the Aq or Bq sensor seems to offer the most advantage. The other sensors and signals would be acceptable in principle, but the issue of reliability makes strain gage systems undesirable. The Aq sensor has the additional advantage of minimum overall system lag, since each of the other signals results to a greater or lesser degree from dynamic response to the forces produced by Aq. This is not important for quasi-static cases such as steady maneuvers or long period gusts, but it could become important in dynamic situations.

A system based on Aq or Bq sensors has, therefore, been chosen for study.

TABLE 3.1. CANDIDATE SIGNALS AND SENSORS FOR LOAD ALLEVIATION SYSTEM

Signal	Sensor	For	Against
Blade Bending Moment	Strain Gage	Senses variable to be controlled	•Questionable reliability •Signal in rotating system •No phase lead
Hub Bending Moment	Strain Gage	Senses variable to be controlled	•Questionable reliability •Signal in rotating system •No phase lead
Aq, Bq Dynamic Pressure Delta Angle of Attack or Sideslip	Pressure Head	•Senses variable which is primary cause of loading •Good reliability •Previous flight experience	
Aircraft Normal, Side Acceleration	Accelerometer	Signal almost in phase with Aq	No use for unaccelerated cases such as unscheduled weight
Wing Bending Moments Torsion, Yaw	Strain Gage	•Sensor in fixed system •Direct measure of variable affecting flying qualities	•Questionable reliability •Lags introduced by wing response •Needs additional sensing to subcontract nacelle moment due to "g"
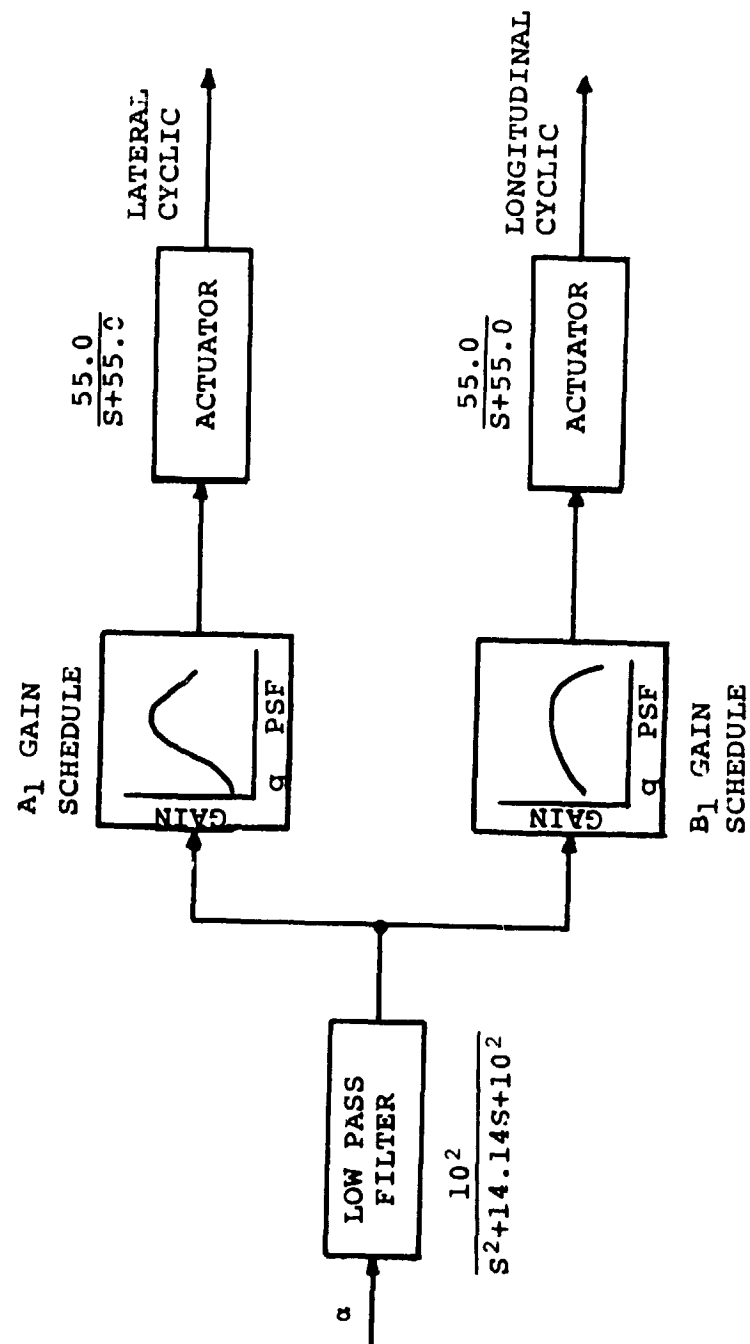
FIGURE 3.1. BLADE LOAD ALLEVIATION SYSTEM SCHEMATIC
OF FEEDBACK LOOP

3.5 SYSTEM CHARACTERISTICS

Figure 3.1 is a schematic of the load alleviation system chosen for study.

The signal sensed is the net increment in angle of attack produced by a gust and the aircraft response. Transfer functions for filters are based on stability considerations and actuator transfer functions are typical of actual hardware. The filter has a cut-off frequency of 10 rad/sec and a damping factor of 0.707. The actuator transfer function is of first order with break frequency 55.0 rad/sec.

3.6 DESIGN OF A SYSTEM EFFECTIVE FOR QUASI-STEADY CONDITIONS

When quasi-steady conditions are considered the decision on system characteristics becomes a matter of:

O Selection of which forces or moments to control; since all hub forces and moments cannot be simultaneously brought to zero, a selection is required.

O How gain and azimuth requirements vary with flight conditions

O What signal shaping (filtering) is required to avoid destabilizing dynamic modes

3.6.1 System Designed to Null Rotor Hub Moments in Cruise

The characteristics of gain and azimuth for a system designed

to work on hub pitching and yawing moments were evaluated. Since only steady state effects are considered, the required A_1 and B_1 gain settings are solvable exactly over a range of flight conditions from knowledge of the rotor hub moment alpha and cyclic derivatives. The results are expressed in terms of azimuth and resultant gain. The azimuth angle is defined as

$$\psi = \text{Tan}^{-1} \left\{ \frac{B_1 \ \text{GAIN}}{A_1 \ \text{GAIN}} \right\}$$

and is a direction perpendicular to the axis about which the swashplate tilts.

Questions to be addressed in this study were:

- O Does system gain and azimuth require scheduling as a function of flight conditions?

- O What is the impact of the system on the hub normal force and moments?

- O What is the impact on aircraft static stability?

The values of A_1 and B_1 gain required were evaluated at different speeds and altitudes from the equations for hub pitch and yaw moments:

$$M_{Y_{TOTAL}} = M_{Y_\alpha}\alpha + M_{Y_{A_1}} A_1 + M_{Y_{B_1}} B_1 = 0$$

$$M_{X_{TOTAL}} = M_{X_\alpha}\alpha + M_{X_{A_1}} A_1 + M_{X_{B_1}} B_1 = 0$$

45
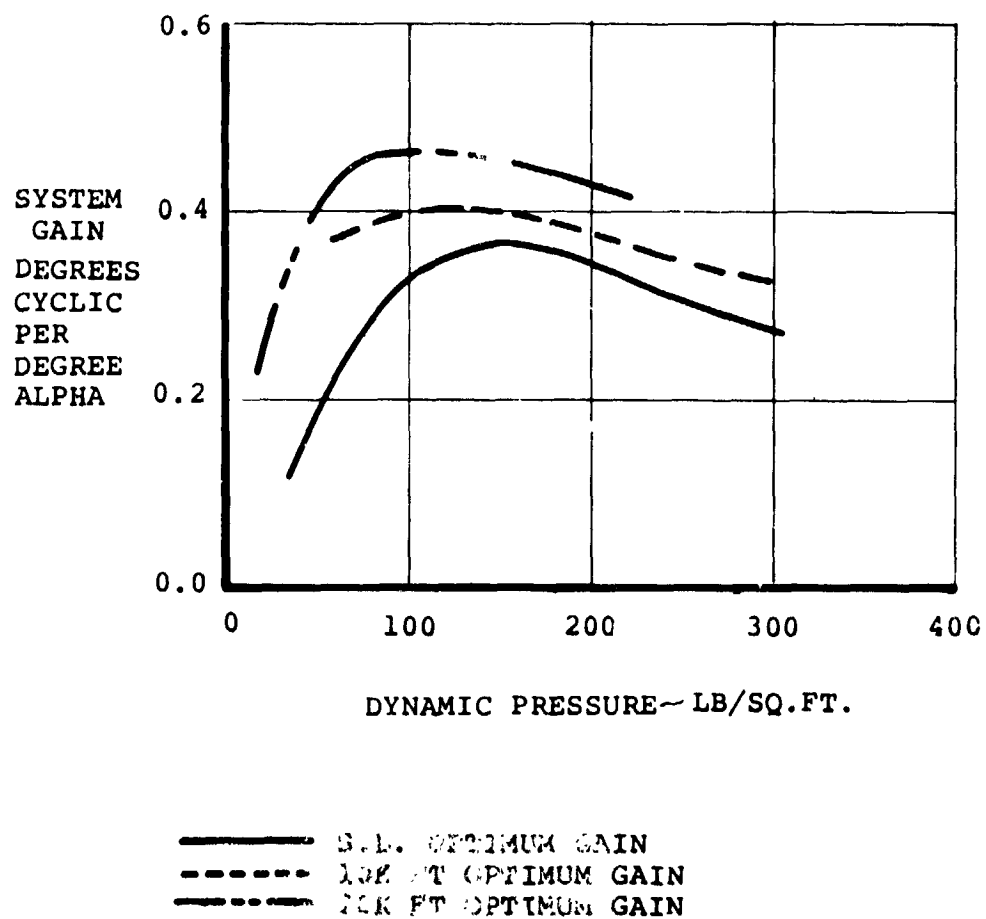
FIGURE 3.2. GAIN REQUIREMENT AS FUNCTION OF DYNAMIC
PRESSURE AND ALTITUDE FOR SYSTEM DESIGNED
TO ZERO-OUT HUB MOMENTS
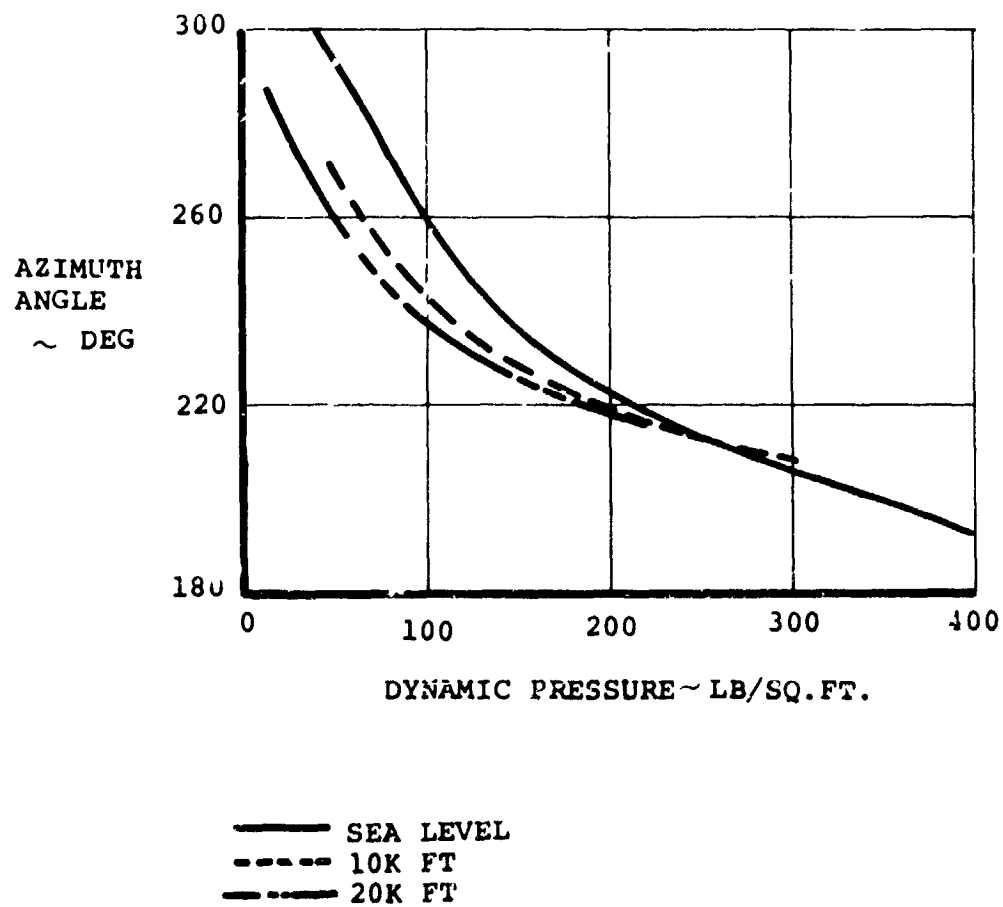
FIGURE 3.3. AZIMUTH ANGLE REQUIREMENT AS FUNCTION OF
DYNAMIC PRESSURE AND ALTITUDE FOR SYSTEMS
DESIGNED TO ZERO-OUT HUB MOMENTS

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These equations are solved for the ratio of A_1 and B_1 to α and to each other and the answers presented in terms of net swashplate cyclic gain and azimuth. Filtering requirements were determined using Bode Diagram Techniques and system stability was confirmed by examination of root locus. The analytical methodology used is incorporated in the C-48 Flying Qualities and Aeroelastic Stability Program. Transient dynamic response was not evaluated for this system.

Figure 3.2 shows the gain required in degrees of cyclic per angle of attack, over a speed range of 100 to 300 knots at altitudes of sea level, 10,000 ft. and 20,000 ft. The associated azimuth angles required are shown in Figure 3.3 and indicate that the angle required drops from around 120° at 100 knots to 30° at 300 knots. The conclusion to be drawn from these curves is that gain and azimuth scheduling as a function of speed is required if the system objectives are to be met at all speeds. The variation with altitude is not so striking so that scheduling of gain and azimuth with altitude is probably not required. The impact of these gain and azimuth settings at sea level on the hub normal and side forces are shown in Figures 3.4 and 3.5. It is seen that normal force and side force derivatives are also reduced by

48

ALTITUDE: SEA LEVEL
Ω = 386 RPM
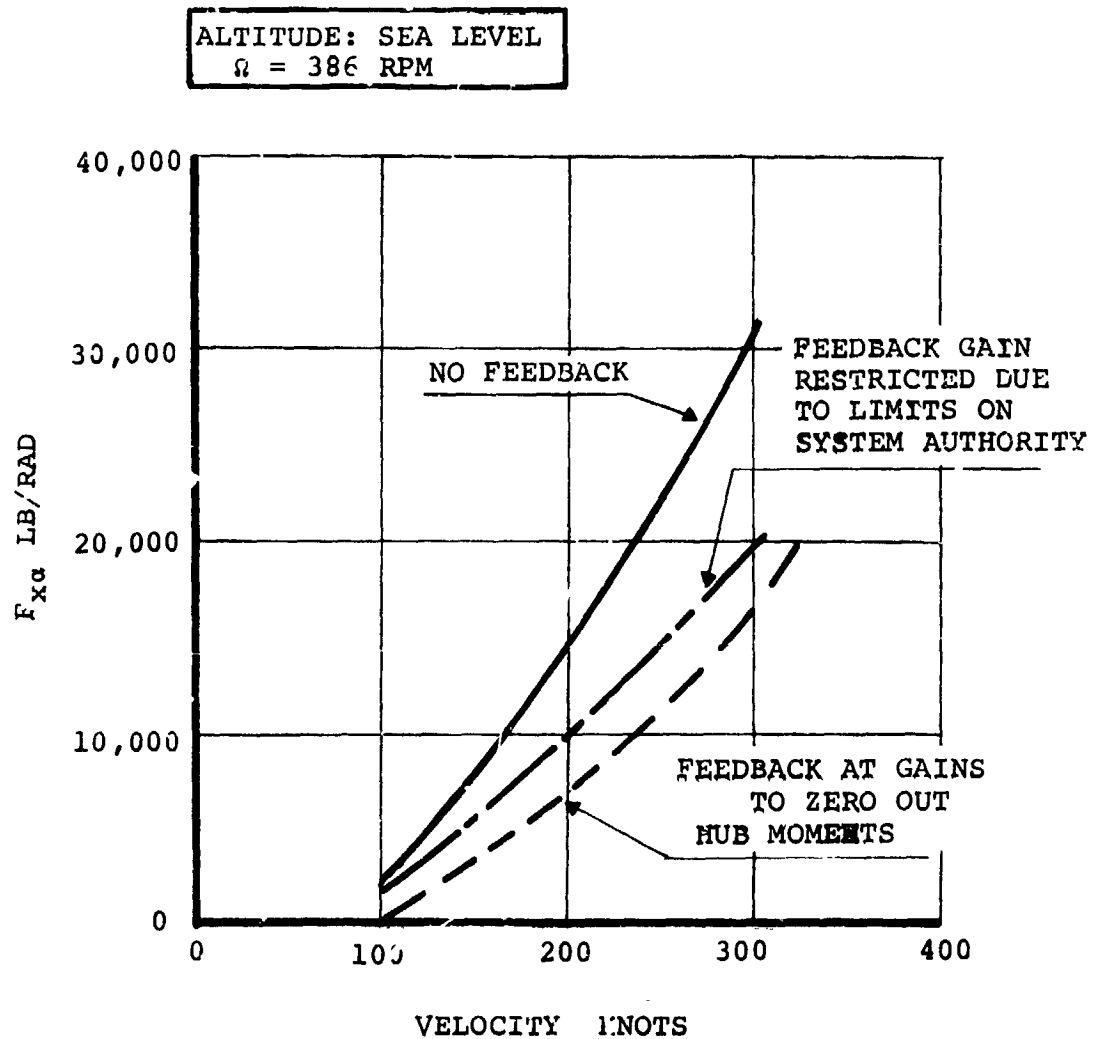
FIGURE 3.4. NORMAL FORCE DERIVATIVE WITHOUT FEEDBACK WITH
UNRESTRICTED GAIN AND WITH ARBITRARY LIMIT OF
1.5° of A_1 and B_1

49
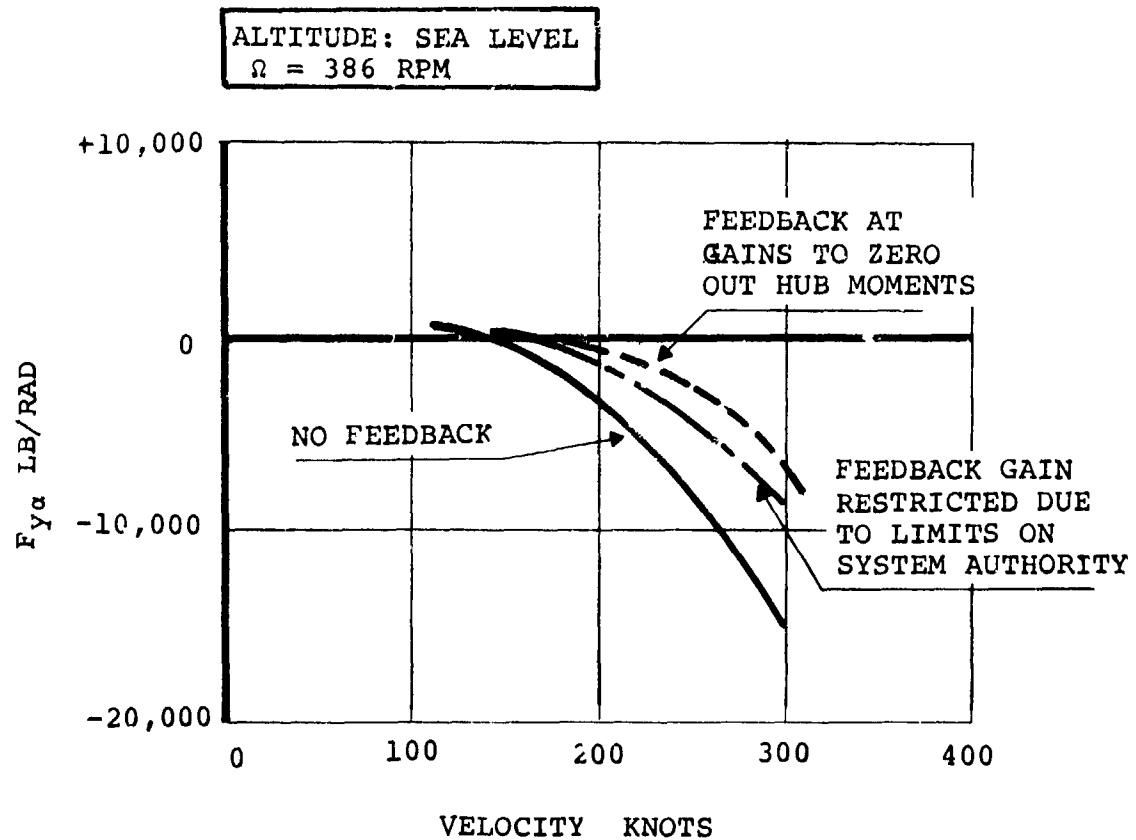
FIGURE 3.5. SIDE FORCE DERIVATIVE WITHOUT FEEDBACK, WITH
UNRESTRICTED GAIN AND WITH ARBITRARY 1.5° LIMIT
ON A_1 and B_1

approximately 50% at the higher speeds.

It is concluded from this study that a system can be defined which will reduce the blade flap bending moments and hub moments to zero, and that the hub normal and side forces will be reduced by the same system. This is a beneficial arrangement for blade loads but may be less acceptable from the point of view of aircraft static stability. The rotor hub pitching moment due to angle of attack is negative, i.e., nose down for low-in-plane stiffness rotors at cruise advance ratios. A reduction of hub pitching moment to zero without a similar reduction in normal force may lead to a net reduction in static margin. That is to say the objective of reduction of blade loads is not necessarily compatible with flying qualities objectives.

3.6.2 System Authority Considerations

Limits may be imposed on the authority of a feedback system because of runaway considerations. That is, unless the system is fail safe which implies triple redundancy, its authority must be less than that available to the pilot or from other control systems at each condition of flight. The stability characteristics of the aircraft will have a discontinuity when the system commands exceed the authority of the feedback

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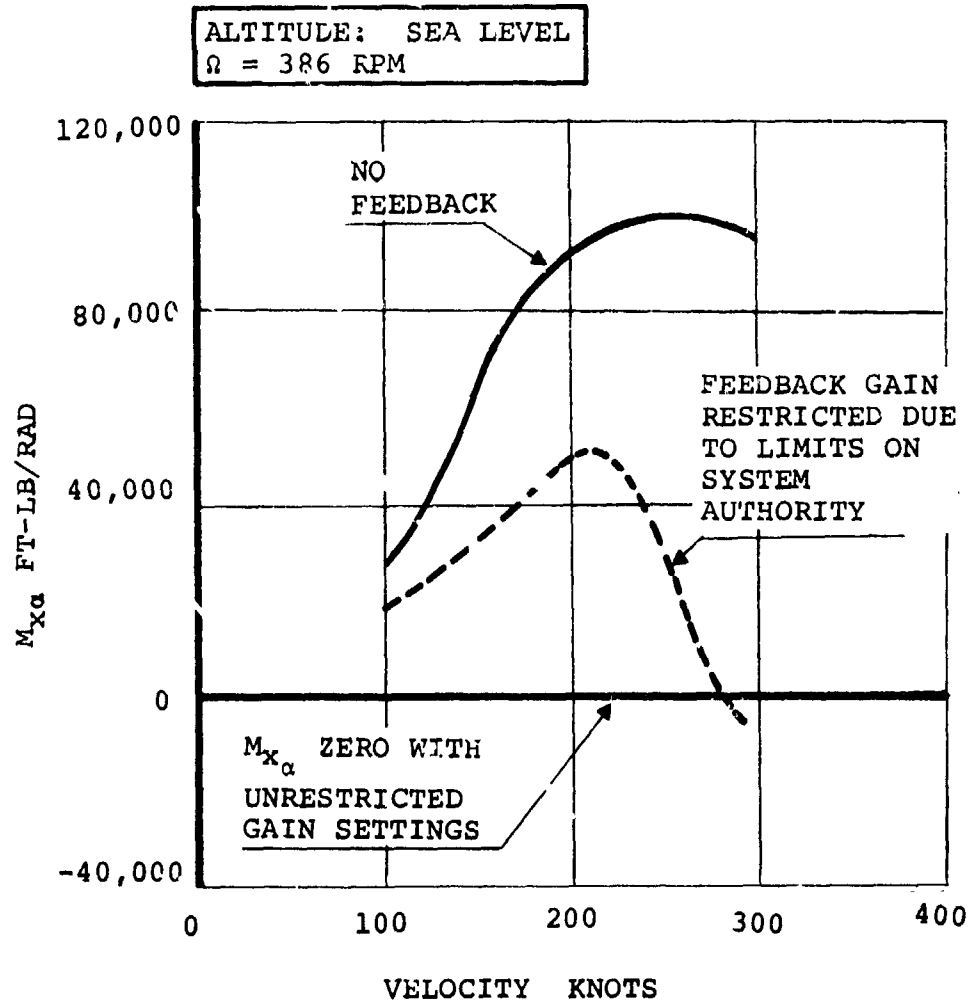
FIGURE 3.6. HUB YAWING MOMENT DERIVATIVE WITHOUT
FEEDBACK AND WITH GAIN RESTRICTED FOR
ARBITRARY LIMIT OF 1.5° A_1 and B_1

52
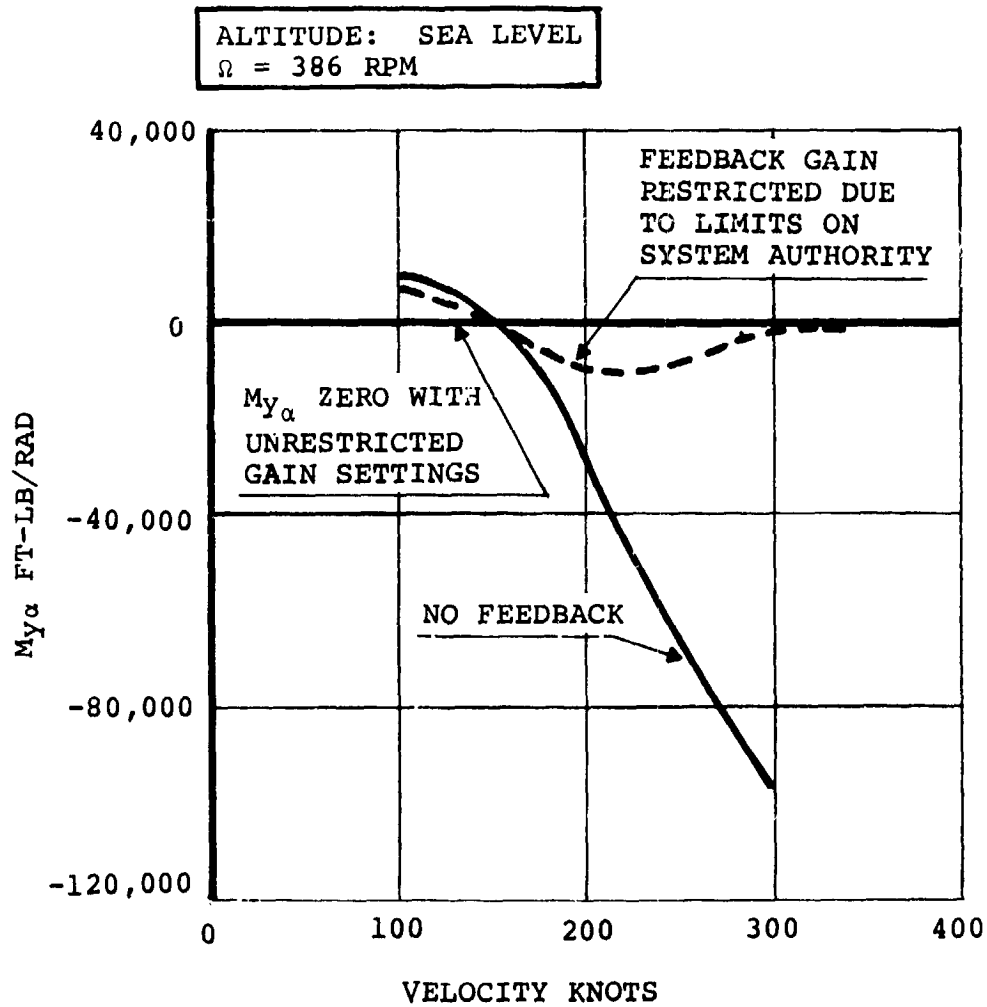
FIGURE 3.7. HUB PITCHING MOMENT DERIVATIVE WITHOUT
FEEDBACK AND WITH GAIN RESTRICTED FOR
ARBITRARY LIMIT OF 1.5° A_1 and B_1

system and since this would be considered unacceptable within

the flight envelope the system gain will be limited so that

flight envelope Aα conditions will not generate demands which

exceed system authority. This places constraint on gain

scheduling which is a function of speed. Figures 3.4 through

3.7 show the impact of gain restrictions set so that an arbi-

trary system authority of 1.5° in the A_1 and B_1 channels is

not exceeded by feedback signal demands associated with maxi-

mum flight envelope conditions. It is noted that even with

restrictions on gain settings there is still a significant

reduction in all the hub forces and moments, reflecting a

similar reduction in blade bending moments and shears. The

net effect on pitching moment about the nacelle pivot is

important in relation to static stability. Figure 3.8 shows

the pivot pitching moment with and without feedback at sea

level and 10,000 ft. At both altitudes the feedback system

reduces pivot pitching moment slightly at low speed thereby

increasing static margin but at high speeds the opposite is

true, with a marked increase in the sea level case. This is

a result of a marked reduction in negative hub pitching

moment which is not accompanied by a similar reduction in

positive normal force.

The net effect on static stability is to provide a slight
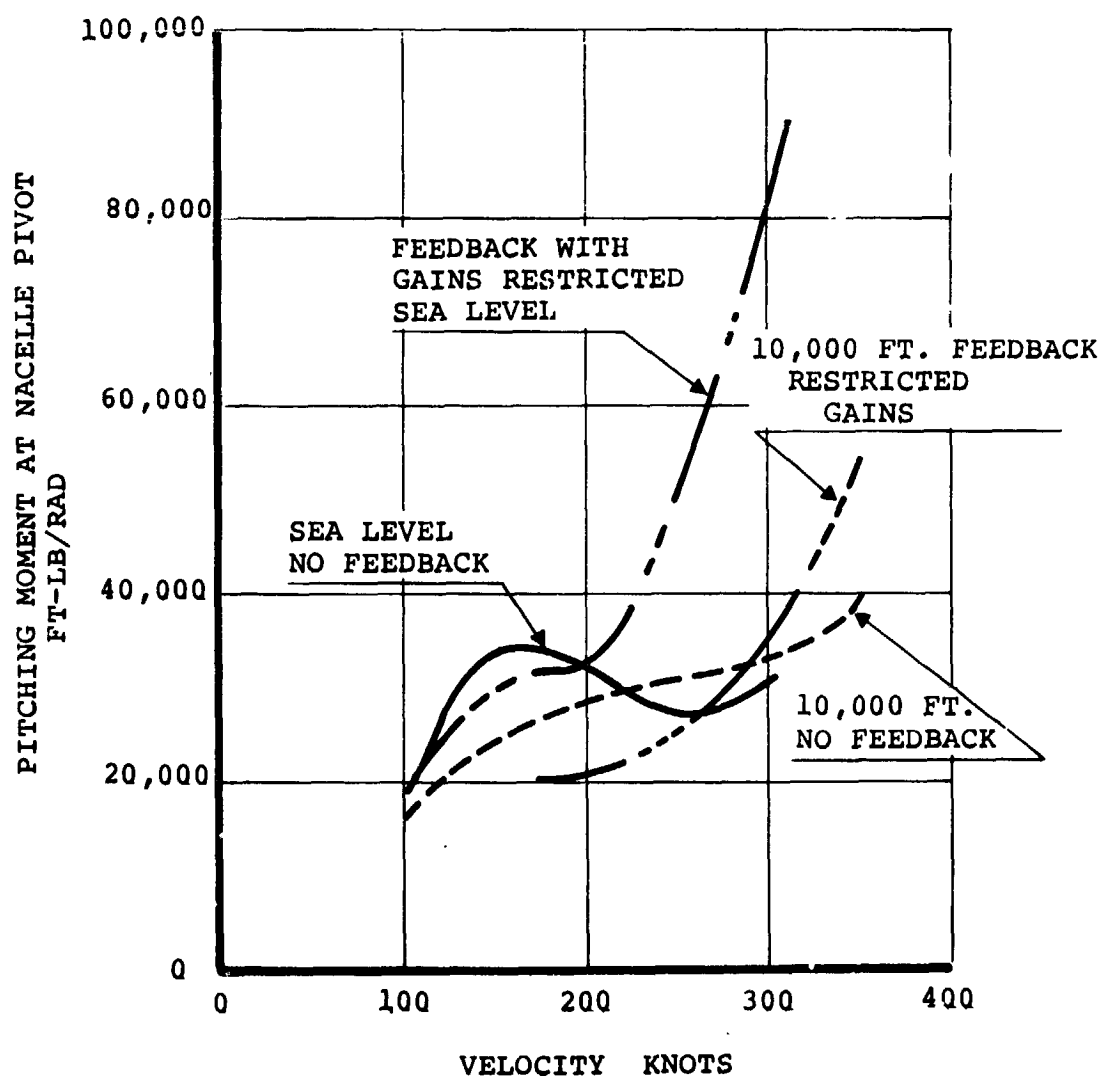
FI GURE 3.8.   PITCHING MOMENT ABOUT NACELLE
              PIVOT WITH AND WITHOUT FEEDBACK

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increase at low speed (1.2% $\bar{c}$ at 150 knots) where improvement
is most useful, and to decrease the static margin by approx-
imately 5% at 300 knots when a decrease is acceptable.

In summary,this system,based on a reduction of hub moments
criterion,also provides reductions in blade loads and normal
side forces, and does not deteriorate the static stability
behavior. However, scheduling of gain and azimuth with speed
is required and preferably with altitude also.

Since the preceeding analysis was based on static considera-
tion only the systems defined were checked for stability by
inspection of their Bode Diagrams. That is the open loop
response of the complete system taking account of blade
dynamics and wing/pylon/fuselage flexibilities and rigid body
freedoms. The diagrams for 150, 250 and 350 knots are shown
in Figures 3.9, 3.10 and 3.11. Decibel levels for 350 knots
are higher than at lower speeds while the phase response is
similar. The levels are for unity gain in the feedback loop.
The net decibel levels are obtained by subtracting the gain
levels indicated. At 350 knots the phase margin is about the
minimum that would be acceptable and a phase shifting network
is indicated to improve this margin.

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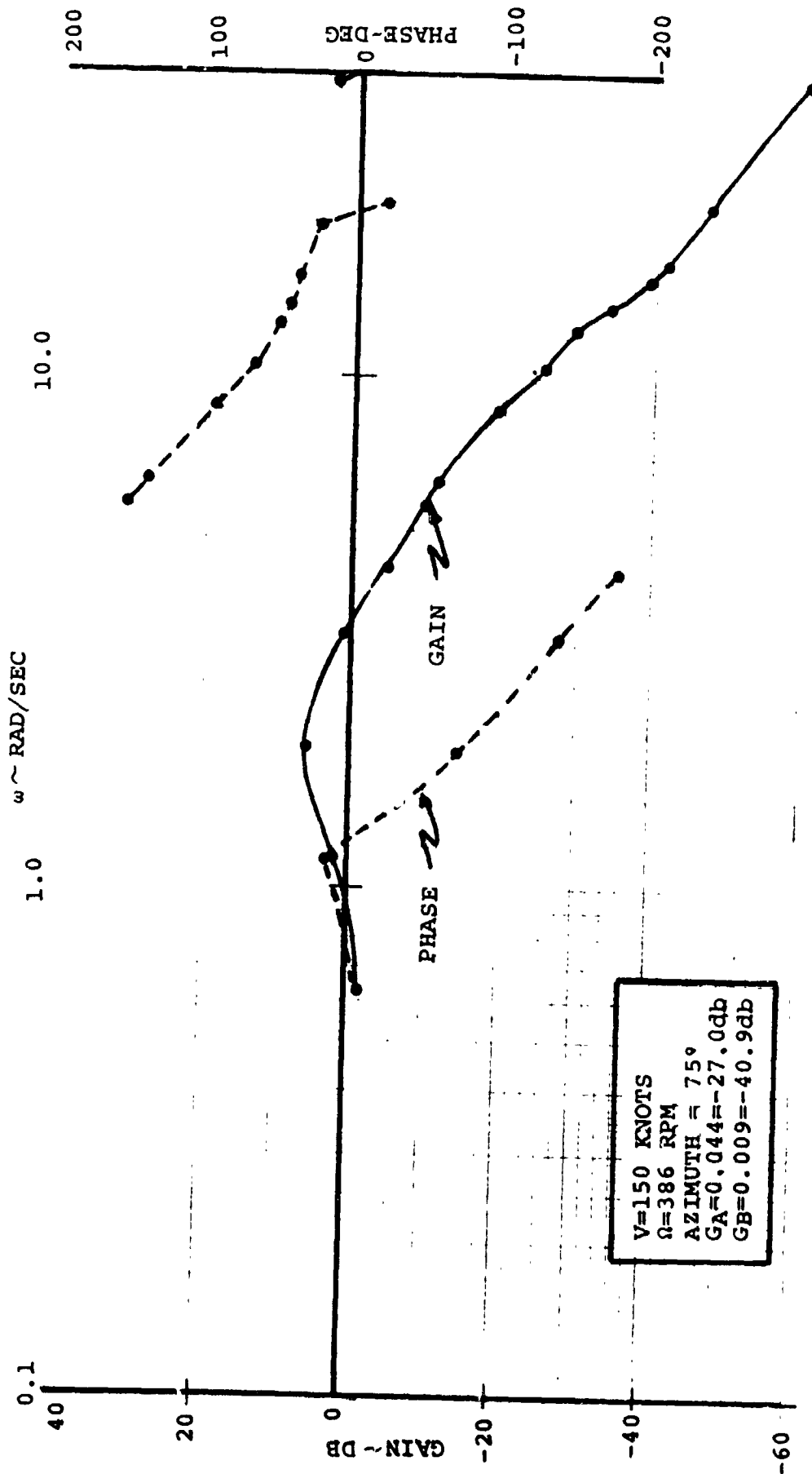
FIGURE 3.9. BODE DIAGRAMS FOR SYSTEM DESIGNED TO REDUCE HUB MOMENTS

V=150 KNOTS
Ω=386 RPM
AZIMUTH = 75°
GA=0.044=-27.0db
GB=0.009=-40.9db

GAIN

PHASE

ω ~ RAD/SEC

PHASE-DEG

GAIN ~ DB
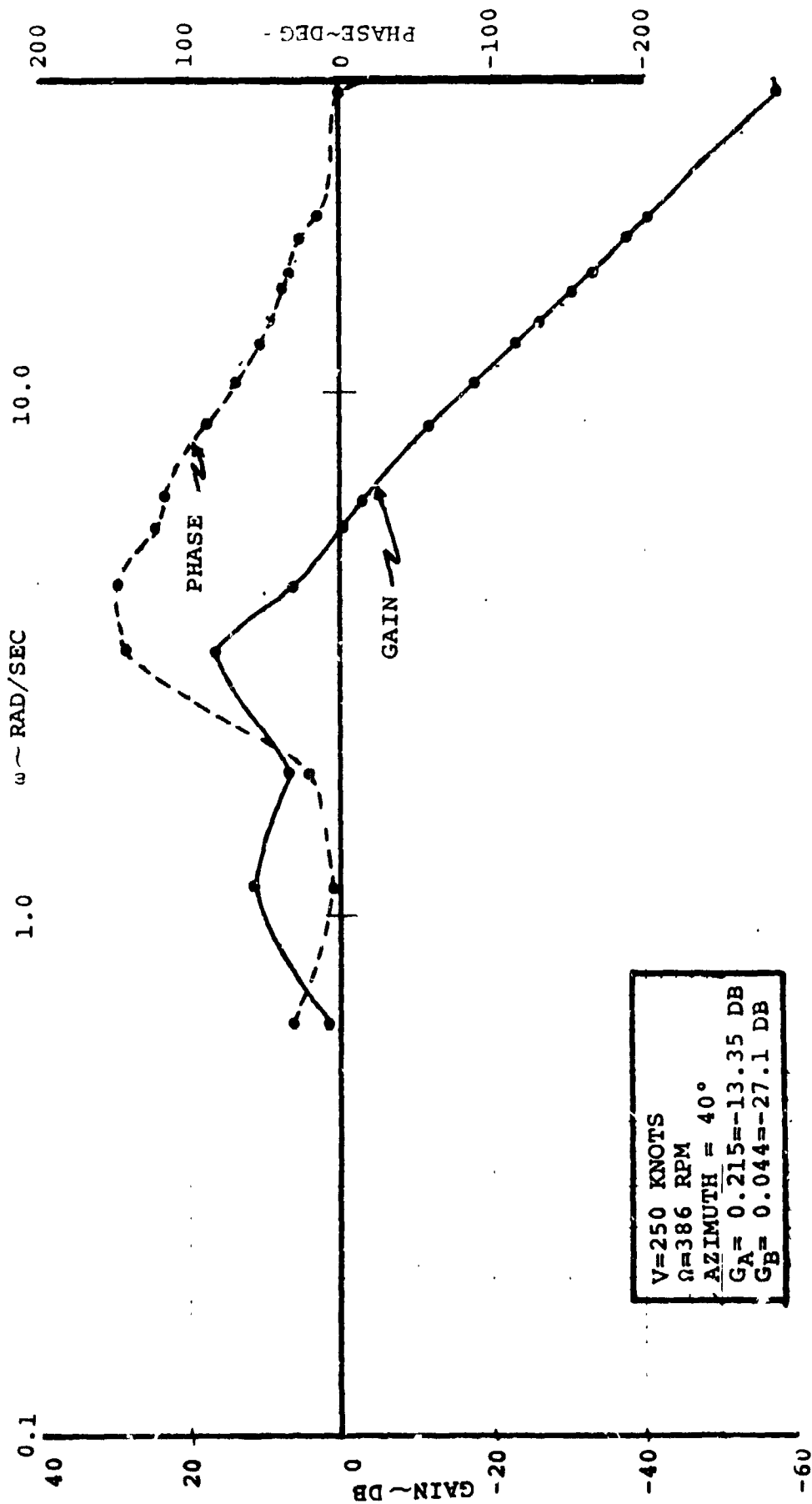
FIGURE 3.10.   BODE DIAGRAM FOR SYSTEM
DESIGNED TO REDUCE HUB MOMENTS

V=250 KNOTS
Ω=386 RPM
AZIMUTH = 40°
G_A= 0.215=-13.35 DB
G_B= 0.044=-27.1 DB
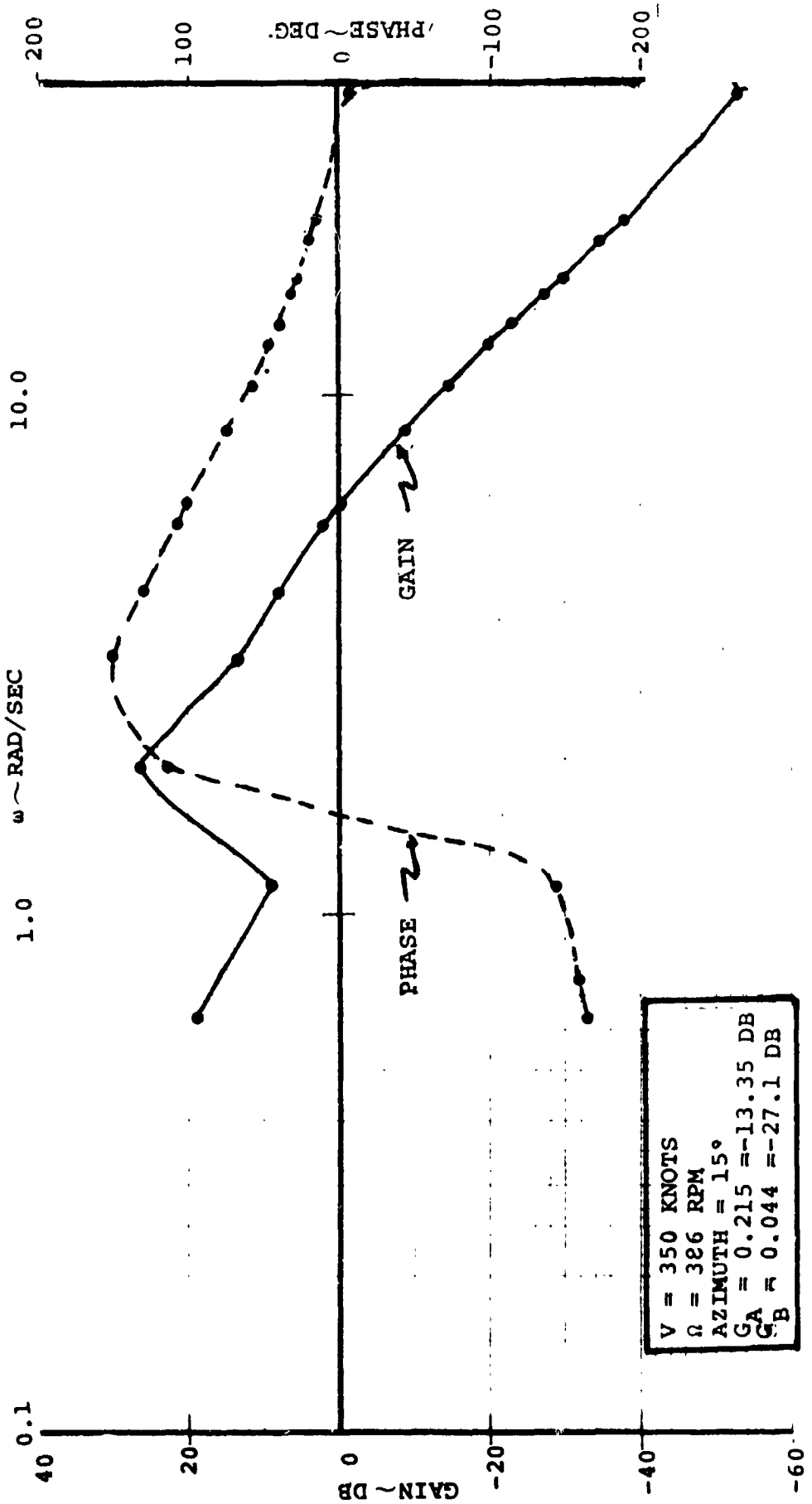
FIGURE 3.11. BODE DIAGRAM FOR SYSTEM DESIGNED TO REDUCE HUB MOMENTS

3.7 ALTERNATIVE SYSTEM DEFINITIONS

Since objectives additional to minimization of hub moments may
be required and since a load alleviation system is required to
operate under transient loading conditions as well as static,
a more general investigation was initiated. In the preceeding
study the system was designed to zero out hub moments due to
steady state loading conditions and it was fortuitous that a
slight improvement in static margin at low speeds came out of
the system. In the present study the behavior of hub forces
and moments and nacelle pivot moments are examined to see if
a better approach is available. To develop a general picture
of the behavior of hub forces and moments and nacelle pivot
moments as functions of gain and azimuth, they were evaluated
for the complete azimuth range and for a set of gain values
ranging from 0 to 1.0 radian of cyclic per radian of shaft
angle. Contours of forces and moments were then plotted as
functions of gain and azimuth as shown in Figures 3.12 and
3.13 for 250 knots and 100 knots respectively. From these
the contours for zero forces and moments and pivot moments
were constructed and superimposed in Figures 3.14 and 3.15.
Examination of Figures 3.12 through 3.15 permits system
parameters to be selected according to different objectives.
For example, if minimization of pivot moments was of
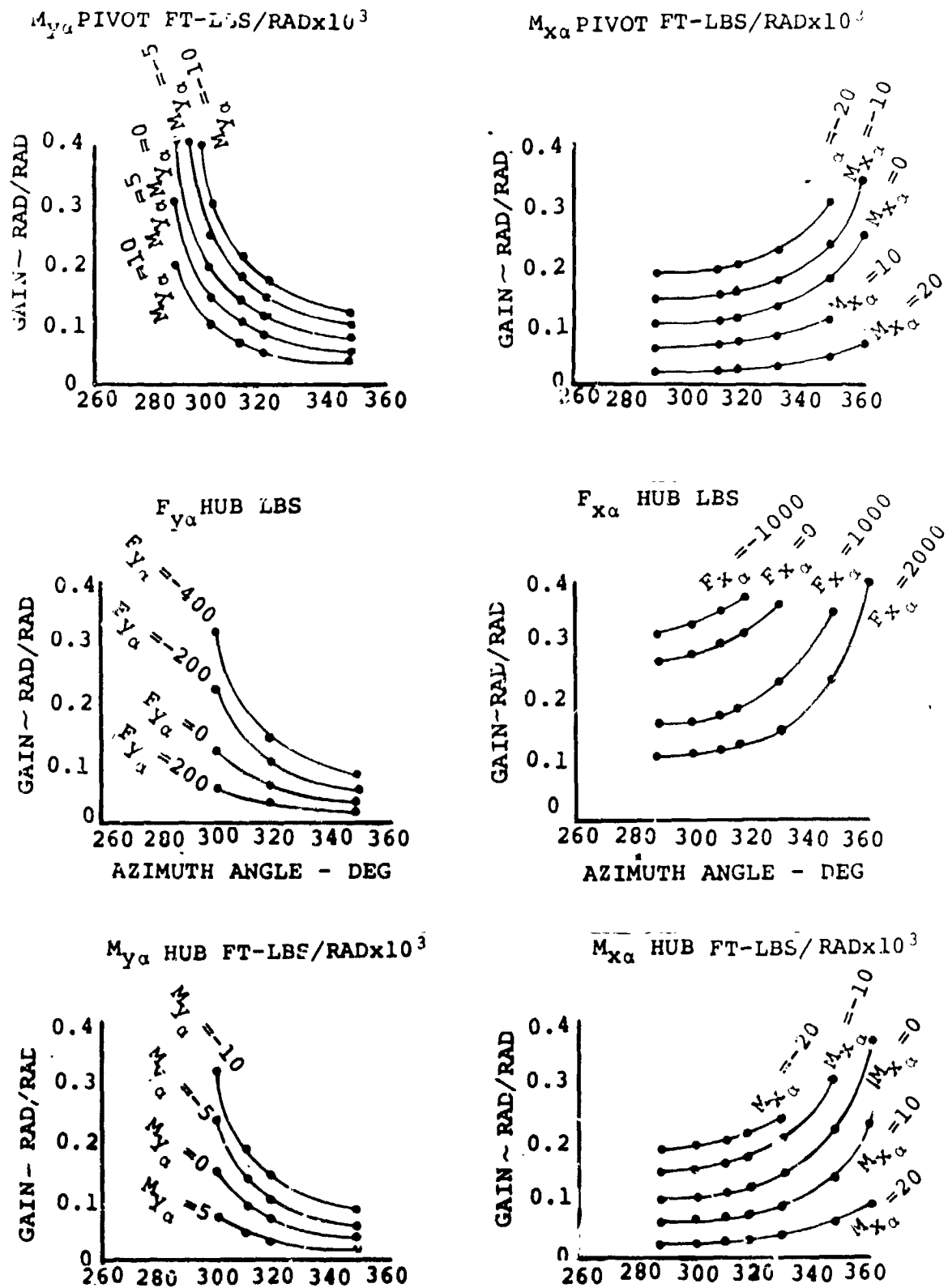
FIGURE 3.12. FORCES AND MOMENTS VS GAIN/AZIMUTH AT 100 KNOTS, 386 RPM AND SEA LEVEL.
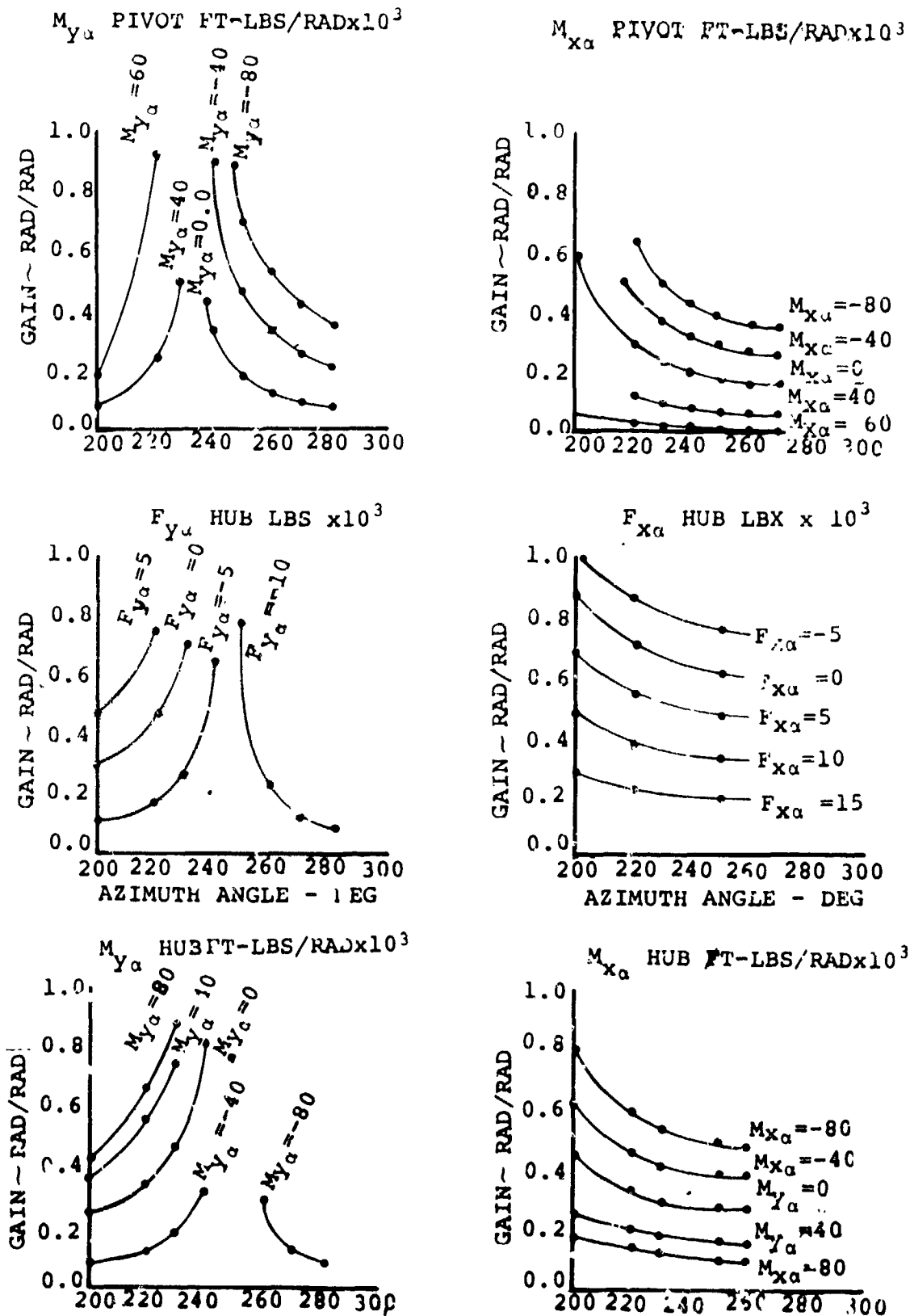
FIGURE 3.13.   FORCES AND MOMENTS VS GAIN/AZIMUTH
AT 250 KNOTS   386 RPM AND SEA LEVEL
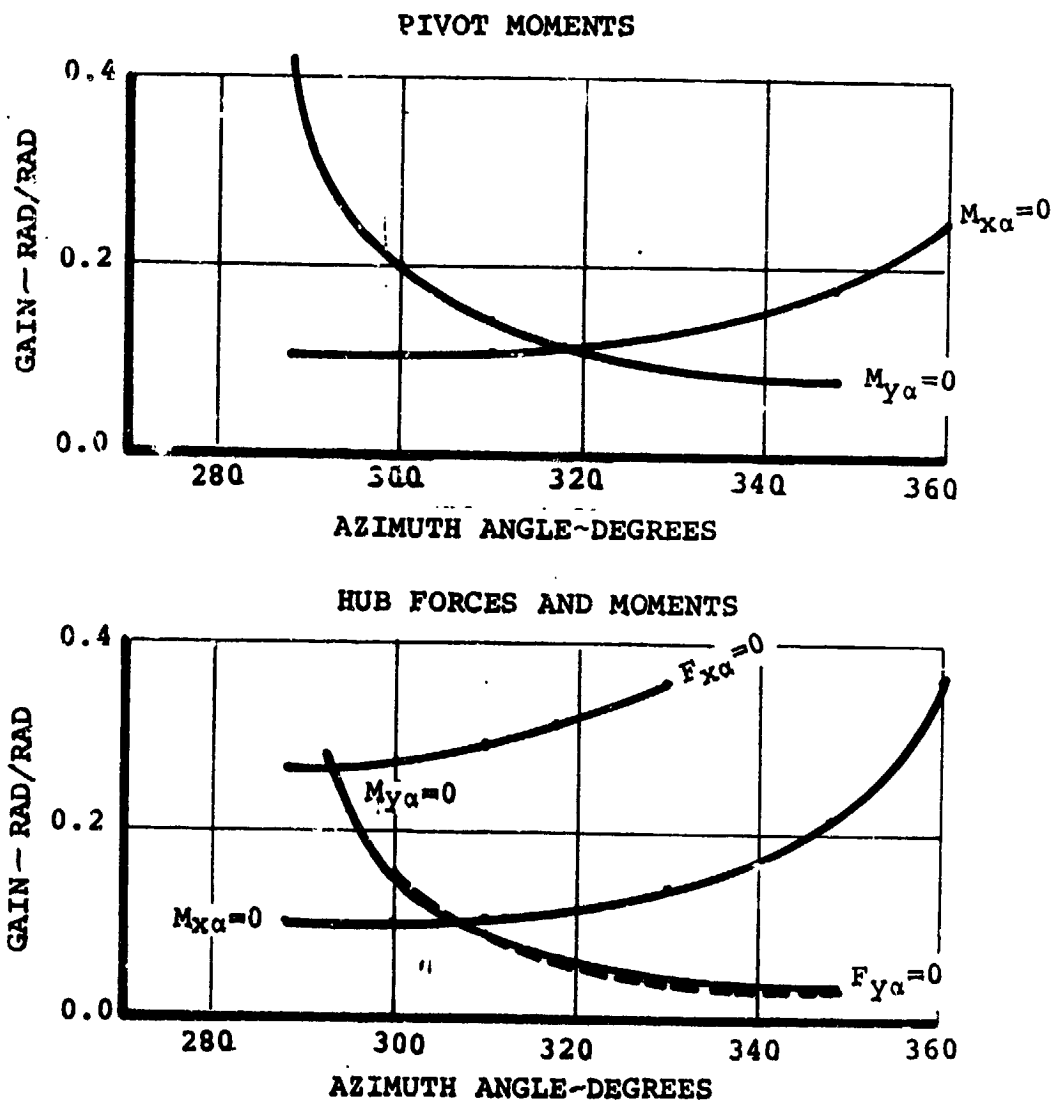
FIGURE 3.14   GAIN/AZIMUTH FOR ZERO FORCES AND MOMENTS
AT 100 KNOTS
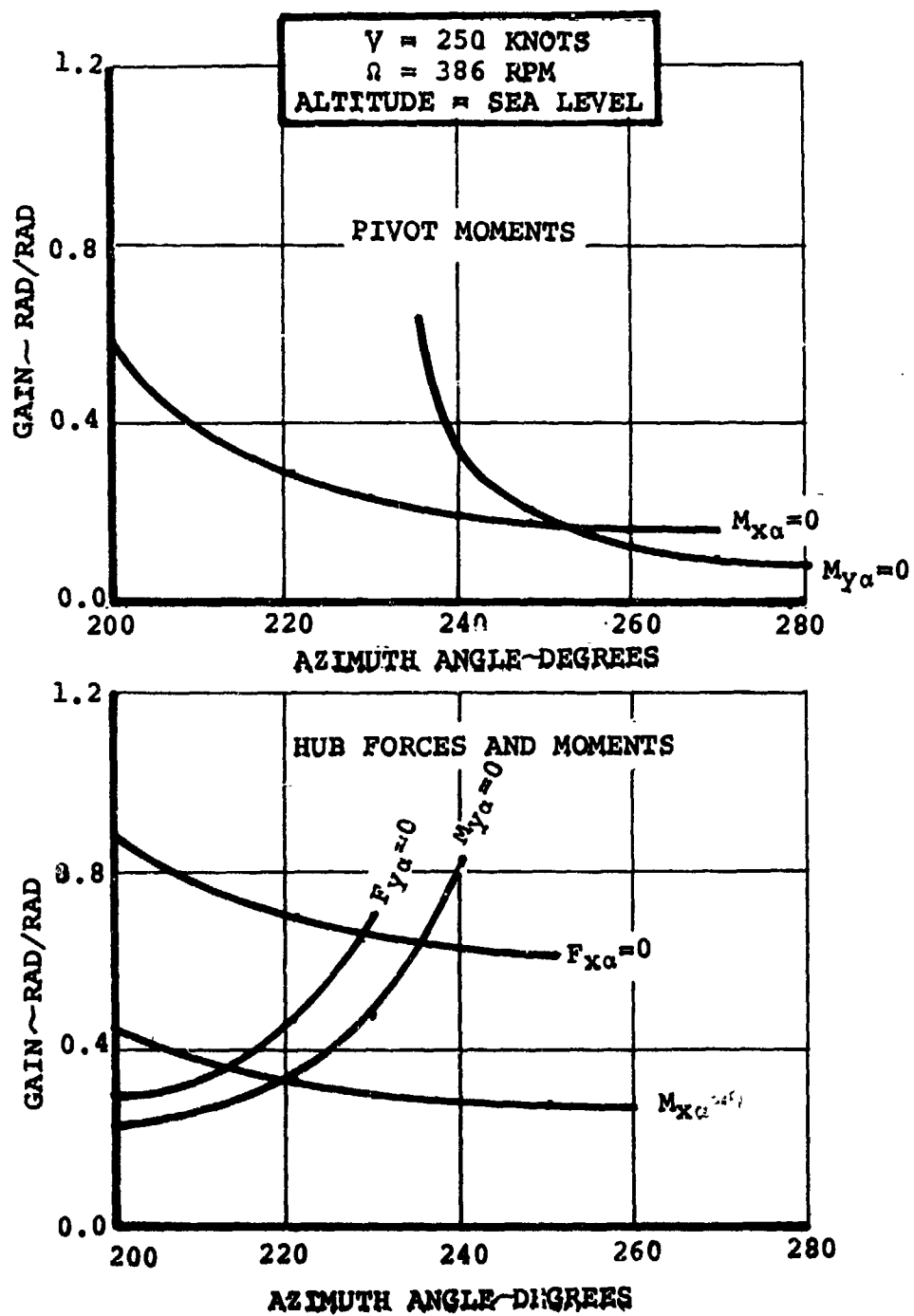
FIGURE 3.15. GAIN/AZIMUTH FOR ZERO FORCES AND MOMENTS
AT 250 KNOTS

D222-10060-3
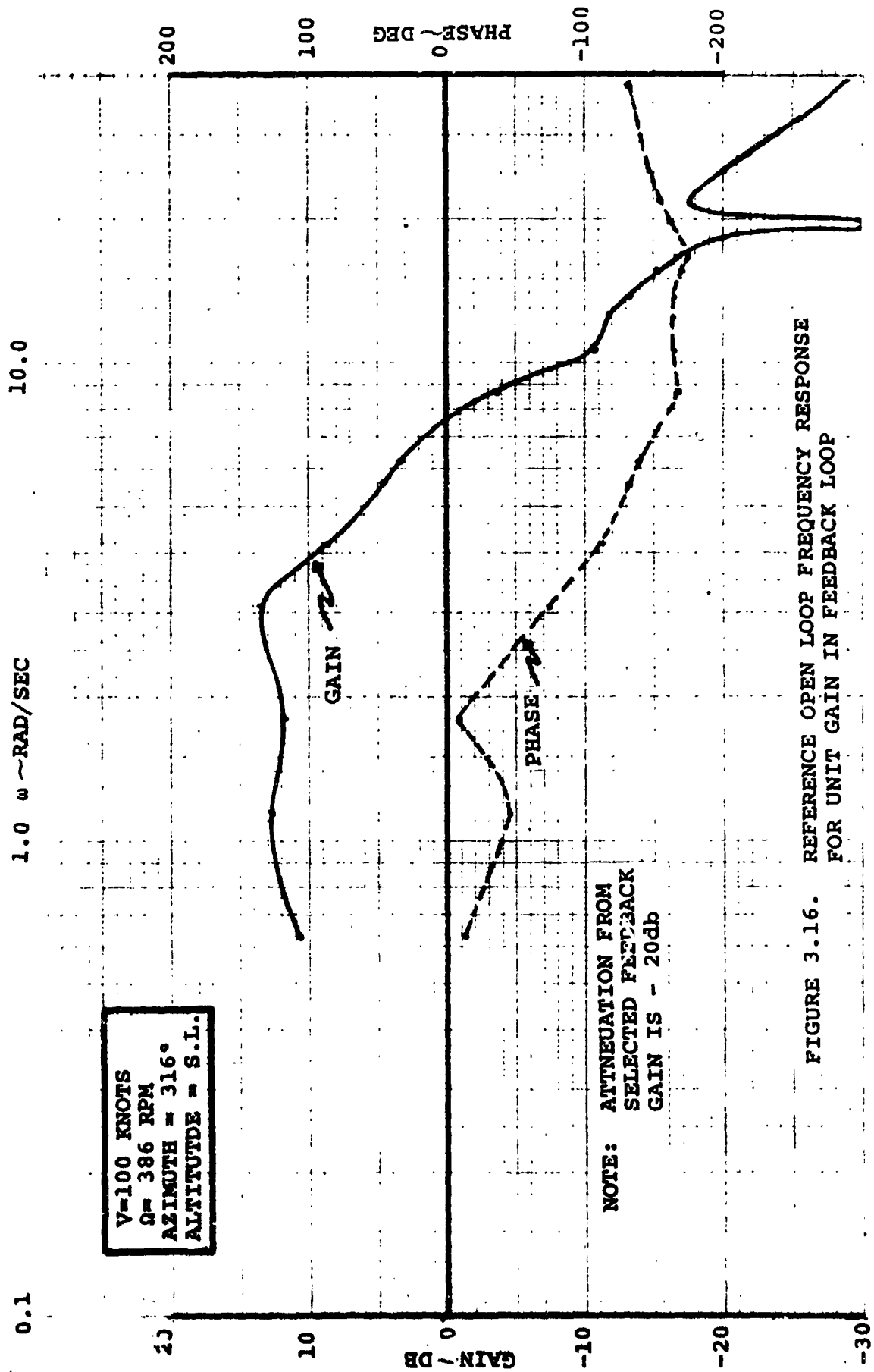
FIGURE 3.16. REFERENCE OPEN LOOP FREQUENCY RESPONSE FOR UNIT GAIN IN FEEDBACK LOOP

65
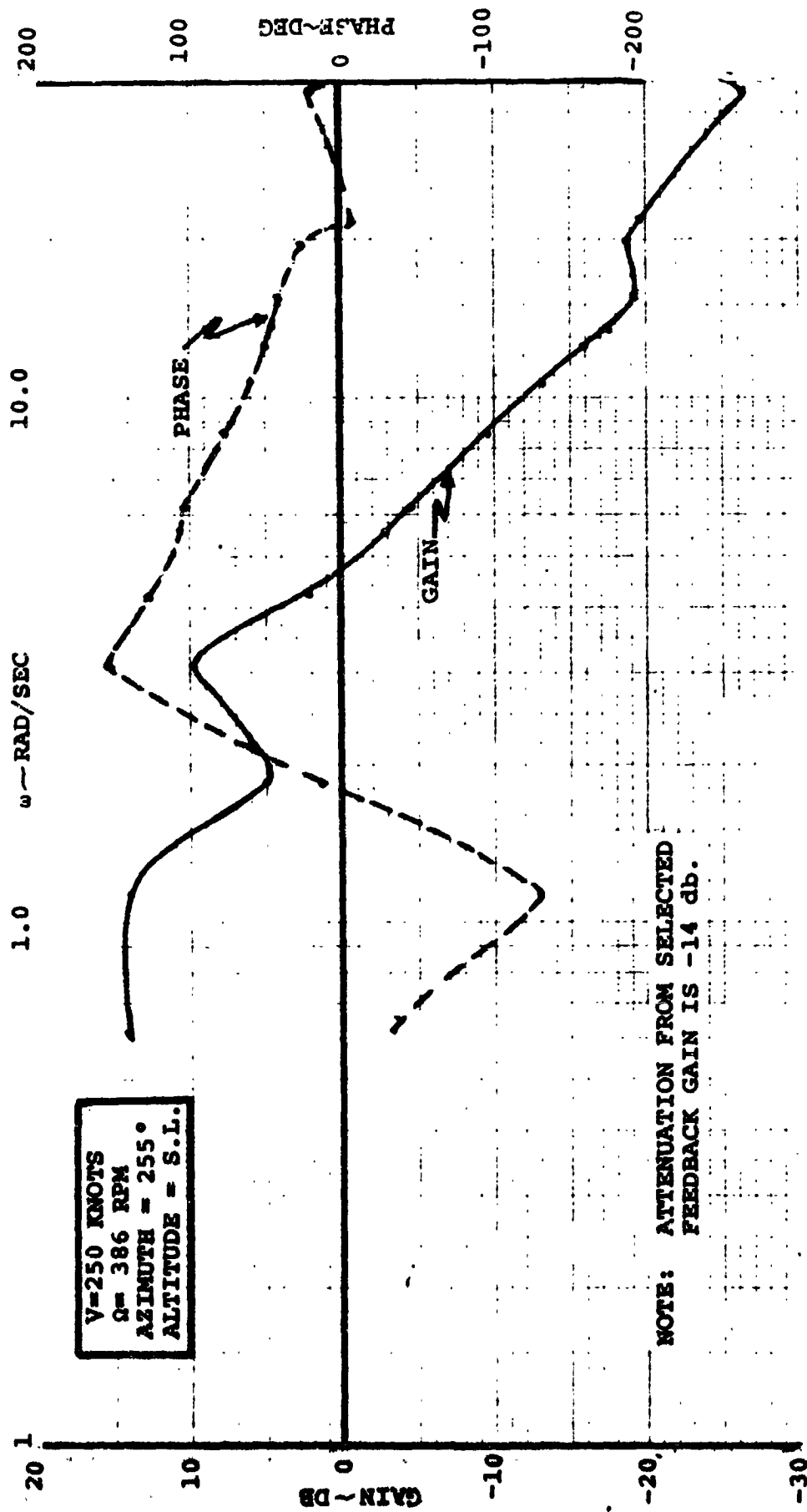
FIGURE 3.17. REFERENCE OPEN LOOP FREQUENCY RSPONSE FOR UNIT GAIN IN FEEDBACK LOOP

NOTE: ATTENUATION FROM SELECTED FEEDBACK GAIN IS -14 db.

V=250 KNOTS
Ω= 386 RPM
AZIMUTH = 255°
ALTITUDE = S.L.

overriding importance, the system gains would be set to give
an azimuth of 255-degrees and a gain of around 0.2 at 250 knots
with scheduling to give an azimuth ot 316-degrees and gain 0.1
at 100 knots. Bode diagrams for these two conditions are
given in Figures 3.16 and 3.17 and it is noted that adequate
gain and phase margins exist. An attractive alternative might
be to reduce the pitching moment about the pivot to zero, and
at the same time minimize hub forces and moments as far as
possible. Thus, by selecting an azimuth around 230 degrees and
gain approximately 0.65, the pivot pitching moment is still
zeroed, but so also are the hub normal and side forces and
pitching moment; only the hub yawing moment remains, and it is
seen from Figure 3.12 that this azimuth and gain setting will
result in a hub yawing moment of approximately -100,000 ft lb/
radian compared with one of approximately +100,000 ft-lb/radian
when no feedback is present. There is, cf course, a net reduction
in total hub moment because the pitching moment has been reduced
to zero. The same reasoning applies at other speeds. At 100
knots the equivalent selection is a gain setting of 0.26 and
azimuth 293-degrees. In this case the total residual hub
moment is approximately -30,000 ft lb/radian compared with a
value without feedback, obtained by resolving 25,000 ft lb/
radian of yawing moment and 10,000 ft lb/radian of pitching
moment.

The above two examples do not exhaust the possibilities; for
example a net nose down pivot moment might be beneficial and

this could be provided by increasing azimuth while keeping a gain setting which made $F_{X\alpha}$ zero.

It is clear that this approach to the selection of feedback system gains and azimuth is a powerful and flexible tool which may be used not only to reduce rotor effects but to actively improve the static stability of the aircraft.

3.8 DYNAMIC RESPONSE CONSIDERATIONS

Many of the situations in which a load alleviation system
might reasonably be expected to work effectively are dynamic
in nature. These include transient maneuvering and gust
loading conditions. In such cases the blade loads and hub
moments depend on the transient response of the blades, which
is governed not only by shaft angle of attack, but by rate
and acceleration associated with the rigid body and flexible
response of the airframe. Thus, in a dynamic situation the
behavior of the whole aircraft must be analyzed with a full
accounting of interactions between gust velocities and rotor,
pylon,wing and fuselage responses.

The potential importance of transient response may be seen
from Table 3.2 in which the frequencies and interactions
associated with the components of the Model 222 Tilt Rotor
Research Aircraft are listed.

The feedback control system envisioned for static cases has
certain fundamental limitations when confronted with the
fully coupled dynamic situation. Ashby's "Law of Requisite
Variety" states that to control a multivariable system, the
number of controls provided must not be less than the number
of independent variables. In the static case there are two
independent variables, blade flap and blade lag, and a good job
of controlling these can be accomplished by the use of the two
parameters, feedback gain and azimuth. In the dynamic situation
there are many more independent degrees of freedom, all in-
fluencing the blade response,to a greater or less degree.

TABLE 3.2. SUMMARY OF FREEDOMS AND INTERACTIONS WITH BLADE FREEDOMS

FREEDOM	FREQUENCY	FORCING FROM GUST	COUPLING TO ROTOR	COUPLING FROM ROTOR
Wing Vertical Bending	3.5 Hz	Direct forcing by wing lift	Acceleration excites $(\Omega - \omega_L)$ mode	$(\Omega - \omega_L)$ response forces wing mode
Wing Chord	5.5 Hz	Small	Yaw excites $(\Omega - \omega_\beta)$ mode	Hub side force and yawing moment excites chord bending
Wing Torsion	6.1 Hz	Small	Shaft angle, position, rate and acceleration excites $(\Omega - \omega_\beta)$	Hub normal force and pitching momen excites wing twist
Aircraft Normal Acceleration	---	Acceleration directly proportional to wing lift	Normal acceleration excites $(\Omega - \omega_\beta)$ mode. Velocity relieves gust velocity.	Hub normal force adds to normal acceleration
Aircraft Short Period Mode	.04 Hz at 80 knots to 0.66 Hz at 250 knots	Forced by lift and pitching moment $\propto$ Vgust and air-craft response	Attitude, rate and acceleration excite $(\Omega - \omega_\beta)$	Hub normal force and pitching momen influence short period characteris tics

NOTE 1. $(\Omega - \omega_L)$ denotes the lower cyclic lead-lag mode $\sim$ 0.25 per rev or 1.6 Hz.

$(\Omega - \omega_\beta)$ denotes the lower cyclic flap mode $\sim$ 0.33 per rev or 2.1 Hz.

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Unless additional feedback loops are provided to give control
over at least those variables which strongly influence the
rotor behavior, swashplate feedback by itself of the type
envisioned in Paragraphs 3.5 and 3.6 will be of limited
effectiveness since it provides control only over a small
component of the total forces acting on the aircraft. Never-
theless useful reduction in blade load transient may be
demonstrated even in the absence of special measures to
control aircraft normal acceleration and short period modal
response.

3.8.1 Method of Investigation and Results. Cruise.
The aircraft was subject to a relatively long period, five
second (1-cosine) gust and the time history of the hub
forces and moments were examined with Aq feedback present at
different azimuths, and a gain setting sufficiently small that
no instabilities were generated. From the variation of forces
and moments with azimuth a region was selected for further
investigation and the variation of loads with gain determined.

The effect of feedback is evaluated from the transient response
of the hub forces and moments. Sample responses are shown in
Figures 3.18 and 3.19 for the 250 knot case with an azimuth setting
of 150-degrees. Two maximum load points in the time history are
observed, an initial loading associated with the peak gust
intensity and an overswing loading of opposite sign. In
some cases the second peak is more severe than the first.

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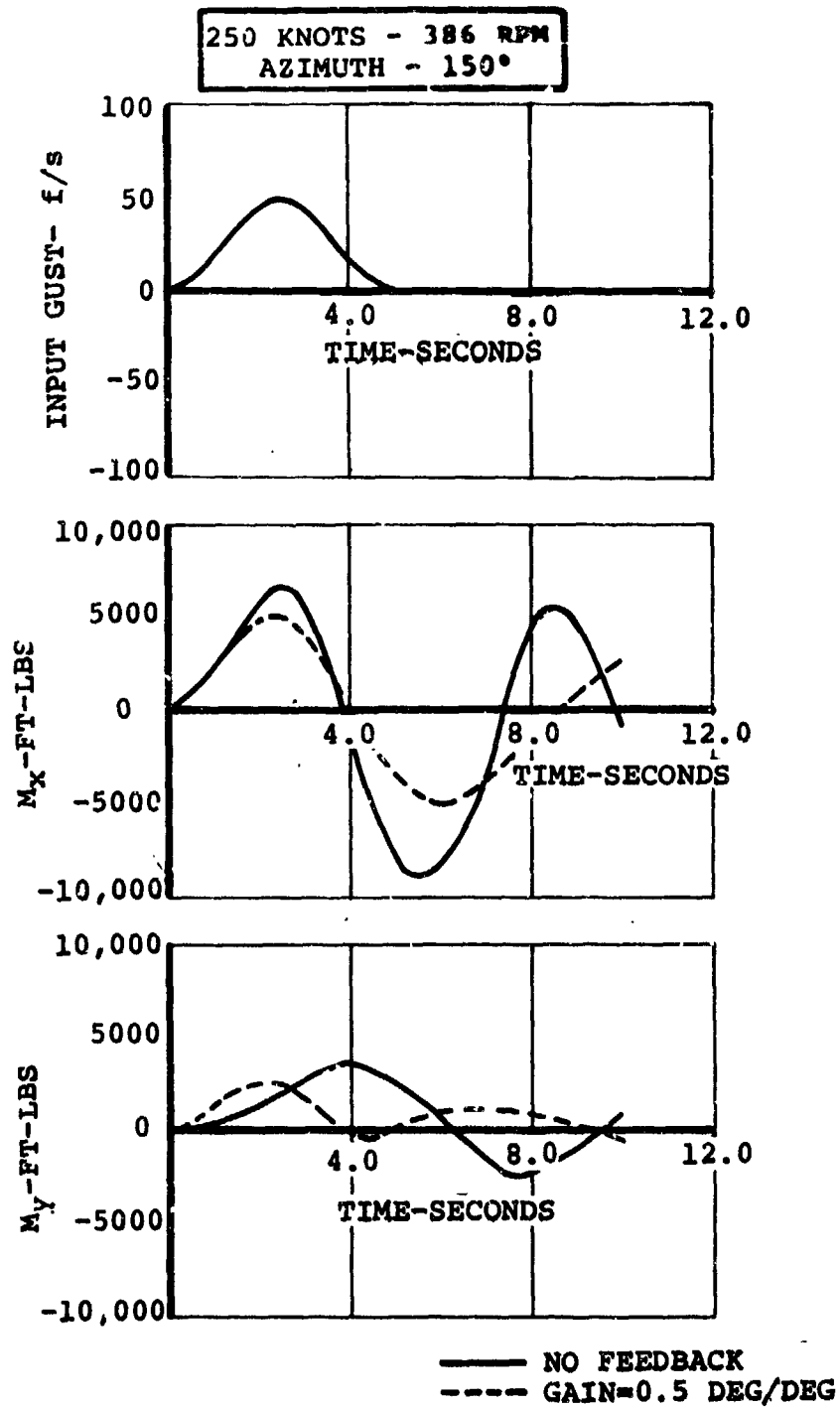
FIGURE 3.18.   EFFECT OF FEEDBACK ON THE DYNAMIC
RESPONSE OF ROTOR HUB MOMENTS, AT
250 KNOTS, 386 RPM.
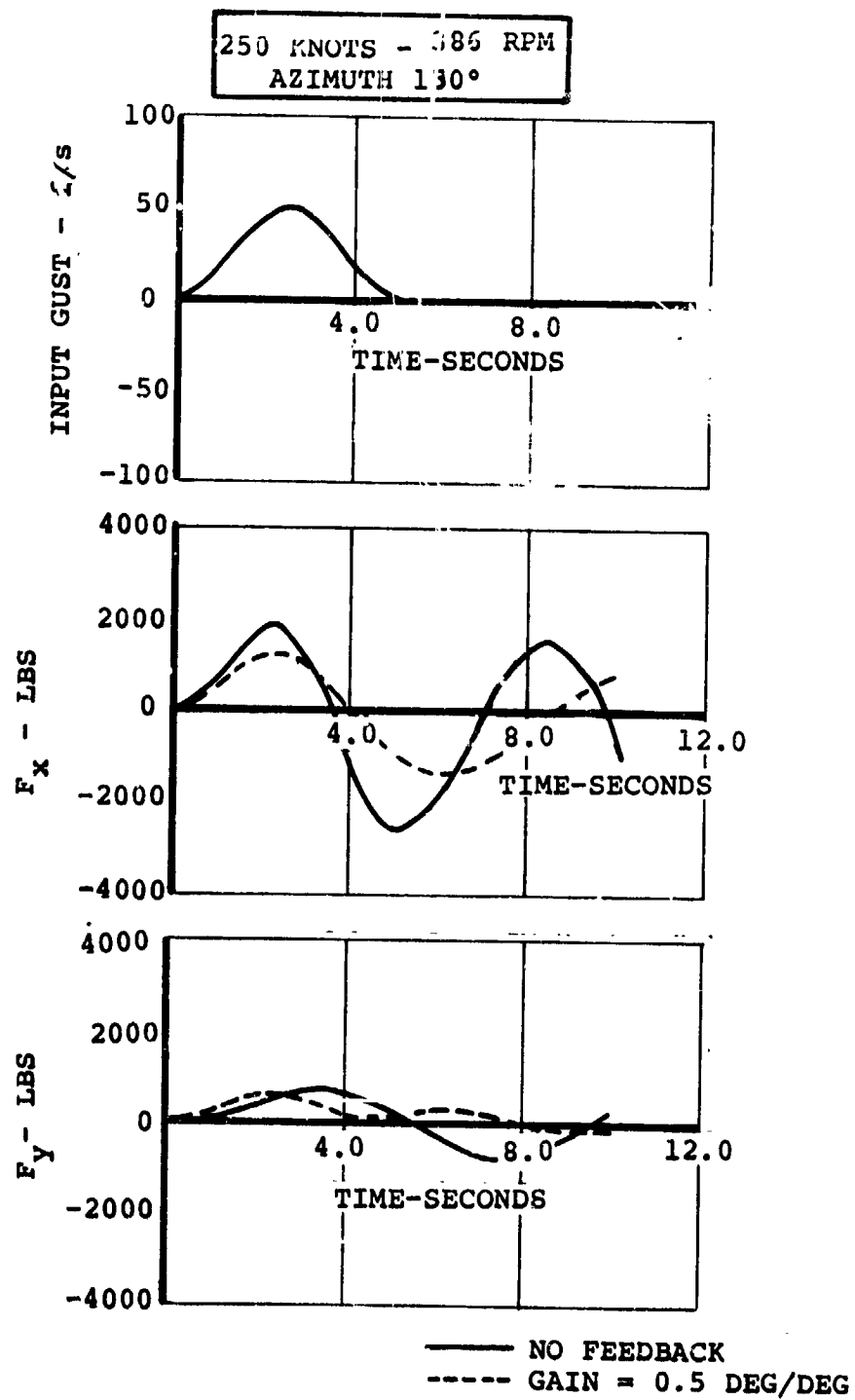
FIGURE 3.19. EFFECT OF FEEDBACK ON THE DYNAMIC
RESPONSE OF ROTOR HUB FORCES AT
250 KNOTS, 386 RPM.

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$$F_x \text{ vs GAIN/AZIMUTH}$$
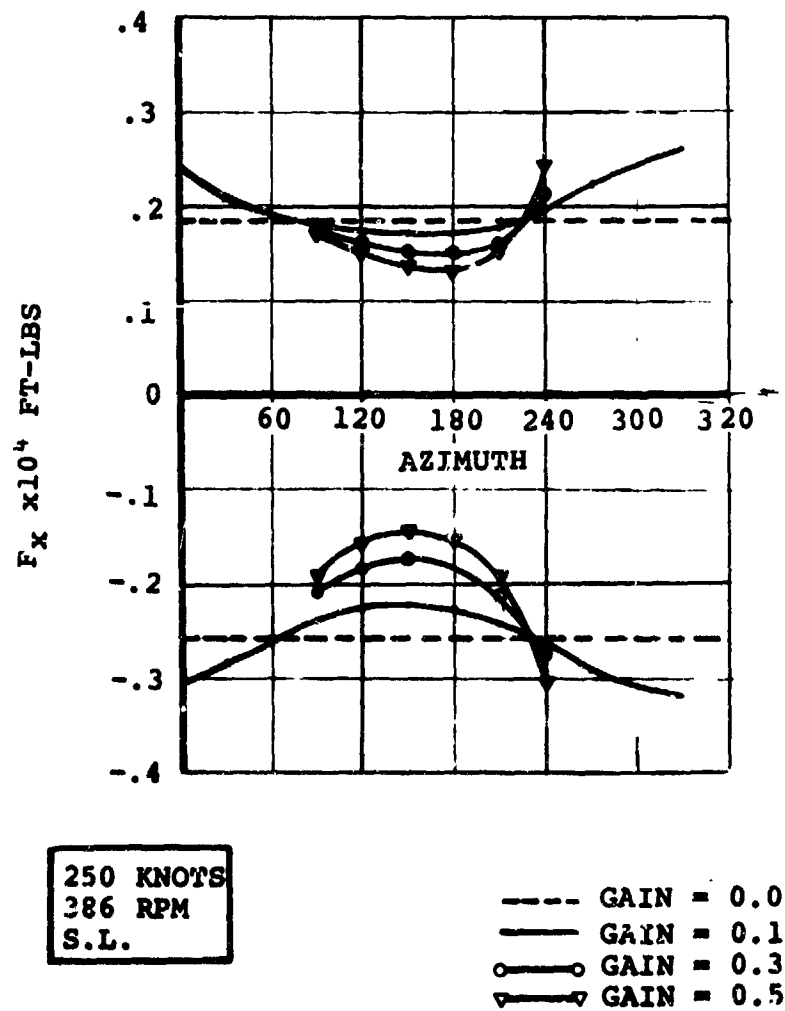
FIGURE 3.20. VARIATION OF PEAK HUB NORMAL FORCE
WITH GAIN AND AZIMUTH OF FEEDBACK

F_y vs GAIN/AZIMUTH

250 KNOTS
386 RPM
S.L.

---- GAIN = 0.0
——— GAIN = 0.1
o——o GAIN = 0.3
▽——▽ GAIN = 0.5
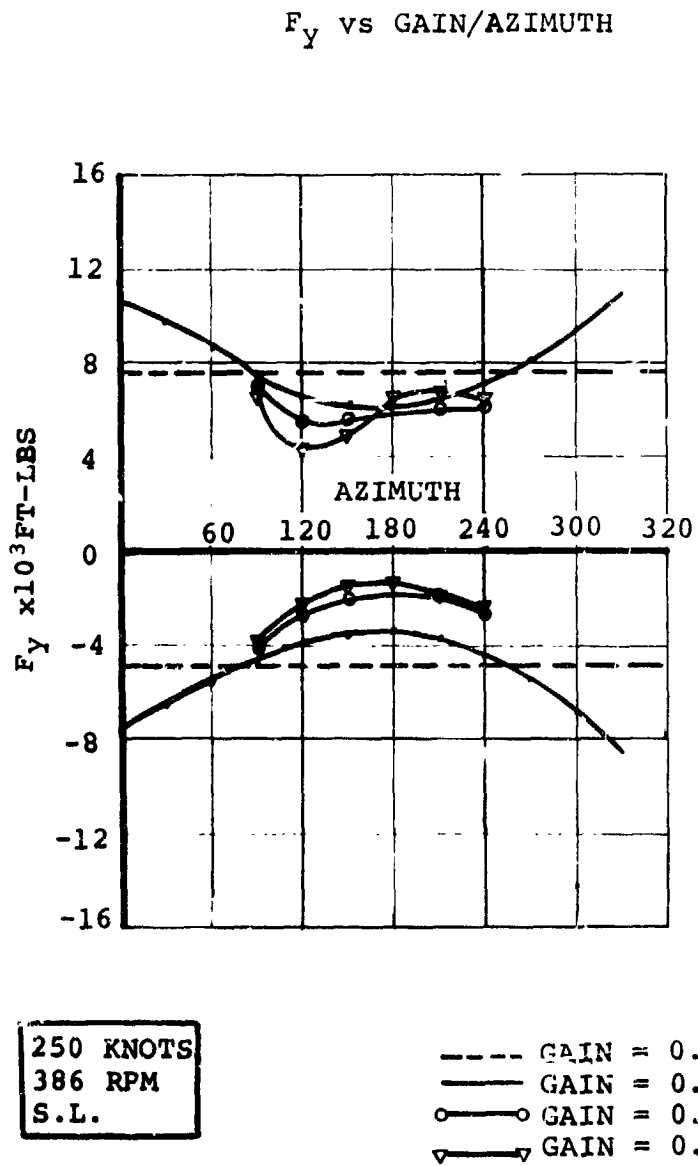
**FIGURE 3.21. VARIATION OF PEAK HUE SIDE FORCE
WITH GAIN AND AZIMUTH OF FEEDBACK**

M_x vs GAIN/AZIMUTH
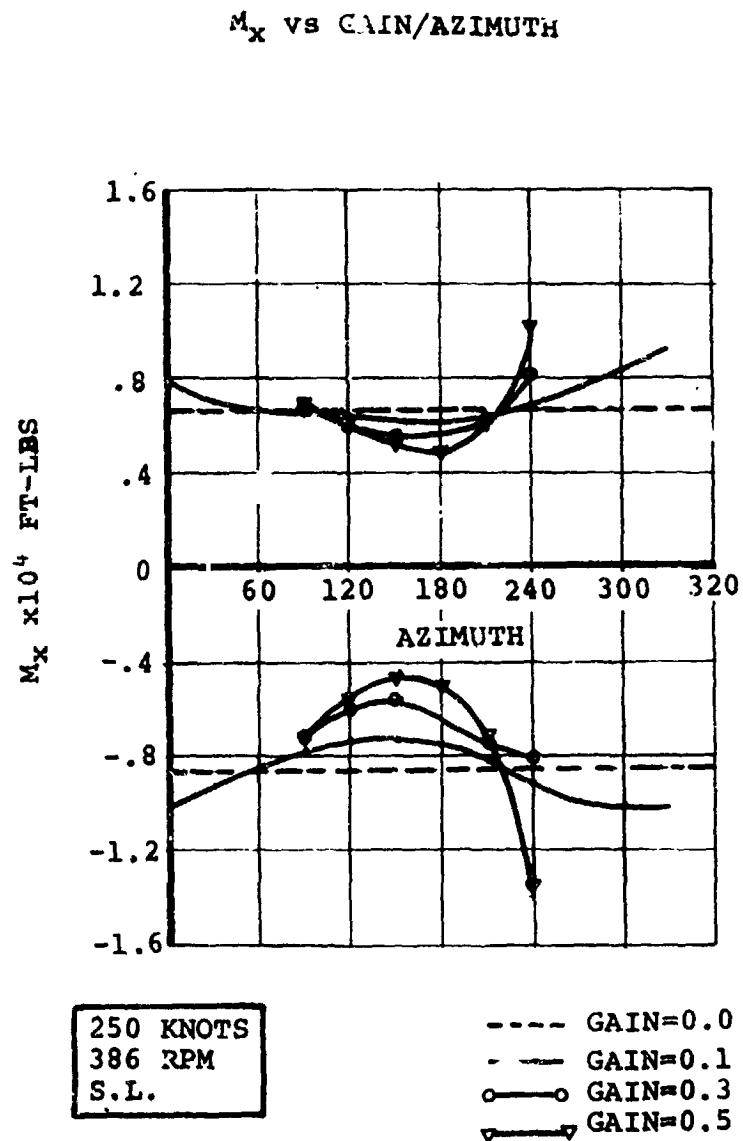
FIGURE 3.22.   VARIATION OF PEAK HUB YAWING MOMENT
WITH GAIN AND AZIMUTH OF FEEDBACK

M_y vs GAIN/AZIMUTH
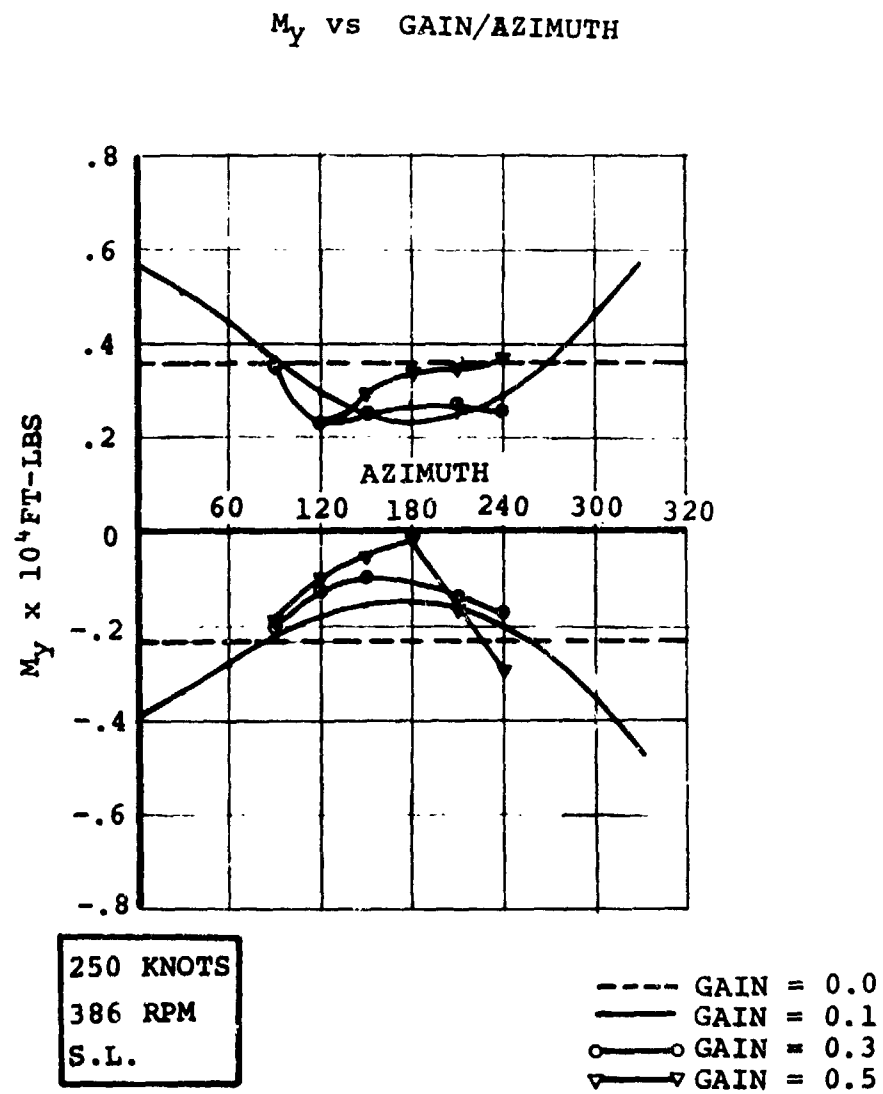
FIGURE 3.23. VARIATION OF PEAK HUB PITCHING MOMENT
WITH GAIN AND AZIMUTH OF FEEDBACK

The peak loads for a range value of azimuth and gain are plotted in Figures 3.20 through 3.23. It is seen that an optimal value of azimuth occurs at approximately 150-degrees, and both the first and second peak loads reduce with gain. The reduction is approximately linear for low values of gain, but a point of diminishing returns is reached around a gain setting of 0.5 degrees per degree of alpha, and at higher settings instabilities were encountered. The reduction in loadings are as follows:

PEAK NUMBER	ZERO GAIN 1	2	GAIN 0.5 1	2	REDUCTION FACTOR 1	2
F_X Normal Force Lb	1850,	-2600	1300,	-1450	0.7	0.55
F_Y Side Force Lb	7600,	-5000	5000,	-1500	0.66	0.3
M_X Yawing Moment Ft-Lb	6600,	-8600	5000,	-4500	0.76	0.525
M_Y Pitching Moment Ft-Lb	3700,	-2300	3000,	-500.0	0.81	0.22

The same system (i.e., gain 0.5 and azimuth 150-degrees) was evaluated for 350 knots as shown in Figure 3.24 and 3.25 with the following results:

PEAK NUMBER	ZERO GAIN 1	2	GAIN 0.5 1	2	REDUCTION FACTOR 1	2
F_X Normal Force Lb	3400,	-3500	1600,	-600	0.47	0.17
F_Y Side Force Lb	1300,	-900	600,	---	0.46	----
M_X Yawing Moment Ft-Lb	5000,	-4500	2500,	-500	0.50	0.11
M_Y Pitching Moment Ft-Lb	2700,	-1800	1500,	---	0.55	----
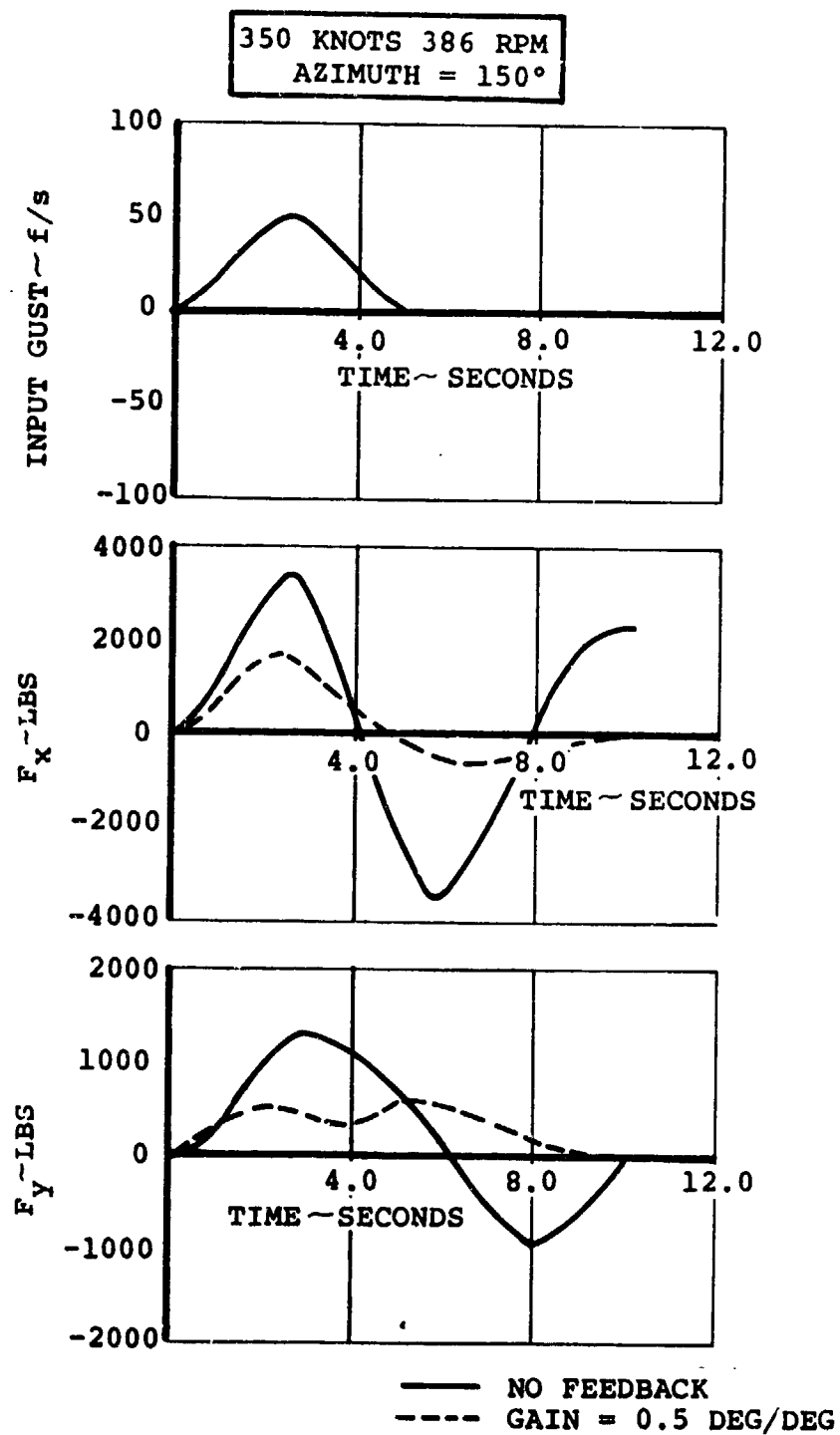
FIGURE 3.24. EFFECT OF FEEDBACK ON THE DYNAMIC
RESPONSE OF ROTOR HUB FORCES AT
350 KNOTS, 386 RPM

79

350 KNOTS 386 RPM
AZIMUTH=150°

NO FEEDBACK
GAIN = 0.5 DEG/DEG
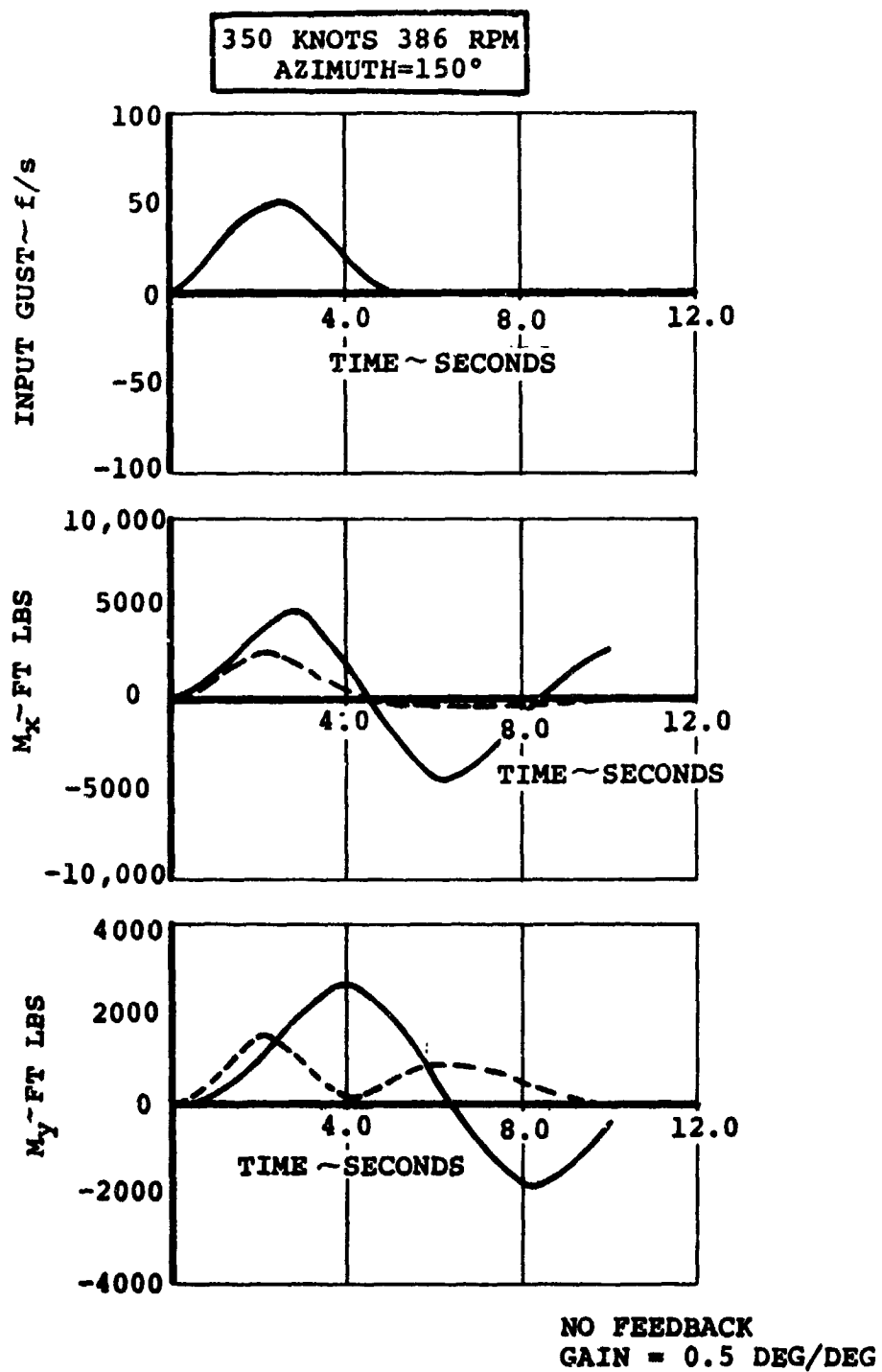
FIGURE 3.25.   EFFECT OF FEEDBACK ON THE DYNAMIC
RESPONSE OF ROTOR HUB MOMENTS AT
350 KNOTS, 386 RPM

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The results presented above indicate that useful reductions in blade load response may be achieved even when swashplate feedback alone is used. However, azimuth settings are significantly different from those determined from purely static considerations although the net gain required is approximately the same.

Attempts to achieve a compromise in azimuth and gain between the static and dynamic cases do not appear to be profitable. As may be seen from Figure 3.15, the azimuth suitable for the dynamic case (150-degrees) is removed from the desirable static value of 216-degrees. Alternatively at the 216-degree value the dynamic system does not produce significant reduction in load. However, the system characteristics may be adjusted with frequency so that such compromises are unnecessary. Since azimuth is a reflection of the relative gain in the A_1 and B_1 channels this affords a method of tailoring azimuth to frequency by the use of different transfer functions for filters in the A_1 and B_1 loops.

3.8.2 Results in Transition Velocities

The effect of azimuth and gain were explored for the transition case of 80 knots and 551 RPM. Figures 3.26 through 3.29 show the variation of forces and moments with azimuth and gain. In this flight condition the most beneficial effects occur around 180-degrees with gain 0.015 degrees per degree of angle of attack. The time history of hub forces and moments is shown

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in Figures 3.30 and 3.31 for this case. The reductions in loads for this case are of a similar order of magnitude to those in cruise, but it is noted that maximum benefit occurs at a different azimuth and there is a substantial increase in the sensitivity to gain.

It is seen that with an azimuth of 180-degrees and a gain setting of 0.15 degrees per degree that the following reductions are obtained.

PEAK NUMBER	ZERO GAIN 1	ZERO GAIN 2	GAIN 0.15 1	GAIN 0.15 2	REDUCTION FACTOR 1	REDUCTION FACTOR 2
F_X Normal Force Lb	550,	---	400,	---	0.73	----
F_Y Side Force Lb	22,	58	----	---	0.0	0.0
M_X Yawing Moment Ft-Lb	10200,	---	9000,	---	0.88	----
M_Y Pitching Moment Ft-Lb	1400,	-3200	600,	-800	0.43	0.25

It is noted that in the transition case the tendency to over-swing loading is significantly reduced. This is a function of the short period mode frequency and damping, which is of the order of 0.04 Hz and 60% critical without feedback, becoming critically damped at the azimuth and gain selected, as shown in Figures 3.32 and 3.33.

3.9 CONCLUSIONS AND DIRECTION FOR DEVELOPMENT

A number of conclusions may be drawn from the preceding studies of load alleviation. The first is that under static conditions, a swashplate feedback system may be designed which will fulfill

F$_X$ VS GAIN/AZIMUTH
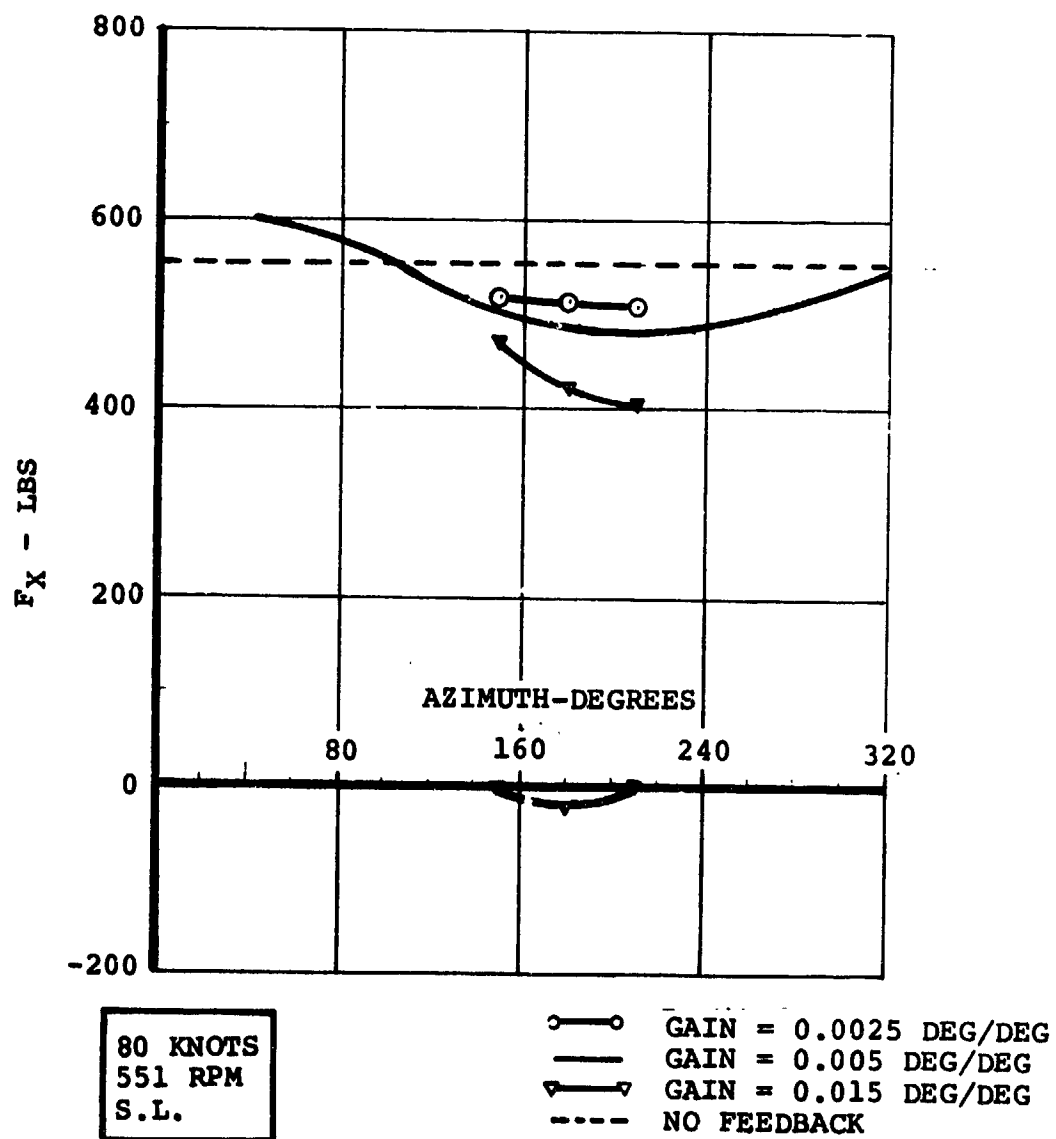
FIGURE 3.26. VARIATION OF HUB NORMAL FORCE WITH
AZIMUTH AND FEEDBACK GAIN AT 80 KNOTS
AND 551 RPM.
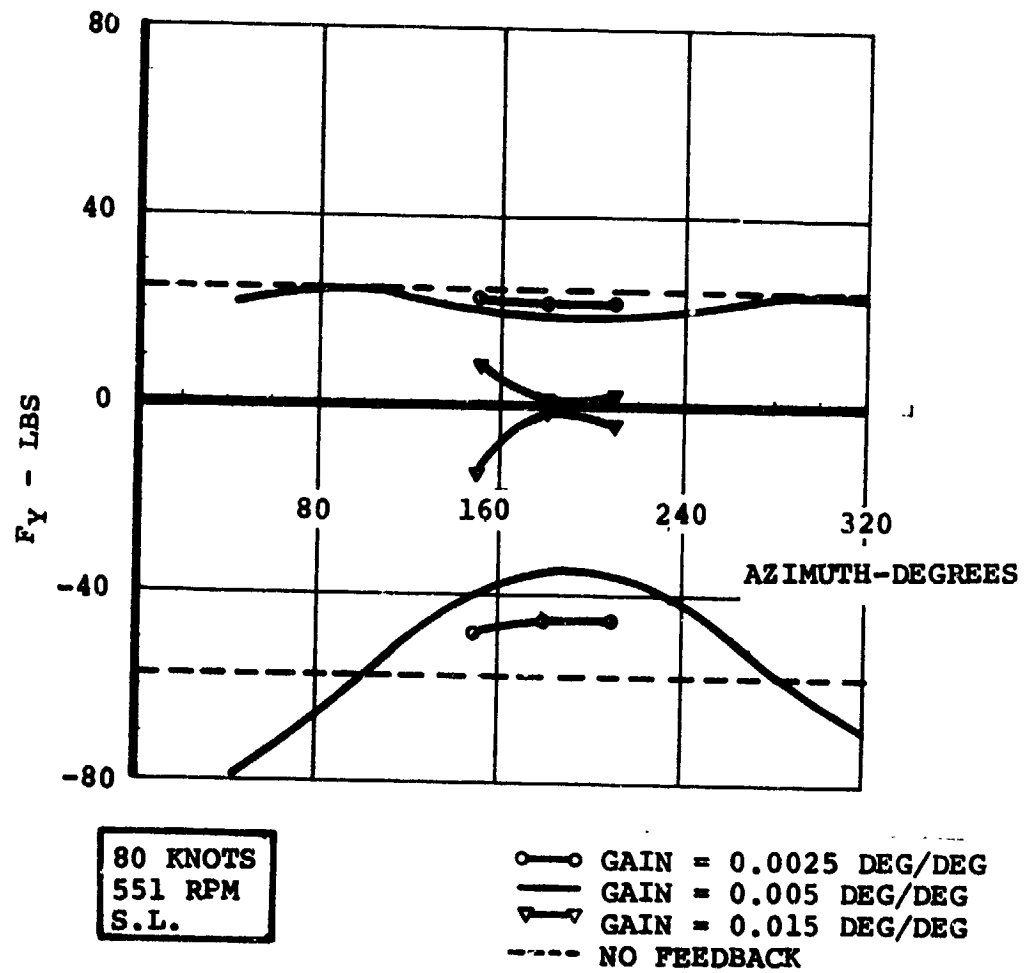
FIGURE 3.27. VARIATION OF HUB SIDE FORCE WITH AZIMUTH AND FEEDBACK GAIN AT 80 KNOTS AND 551 RPM.

M_X VS GAIN/AZIMUTH
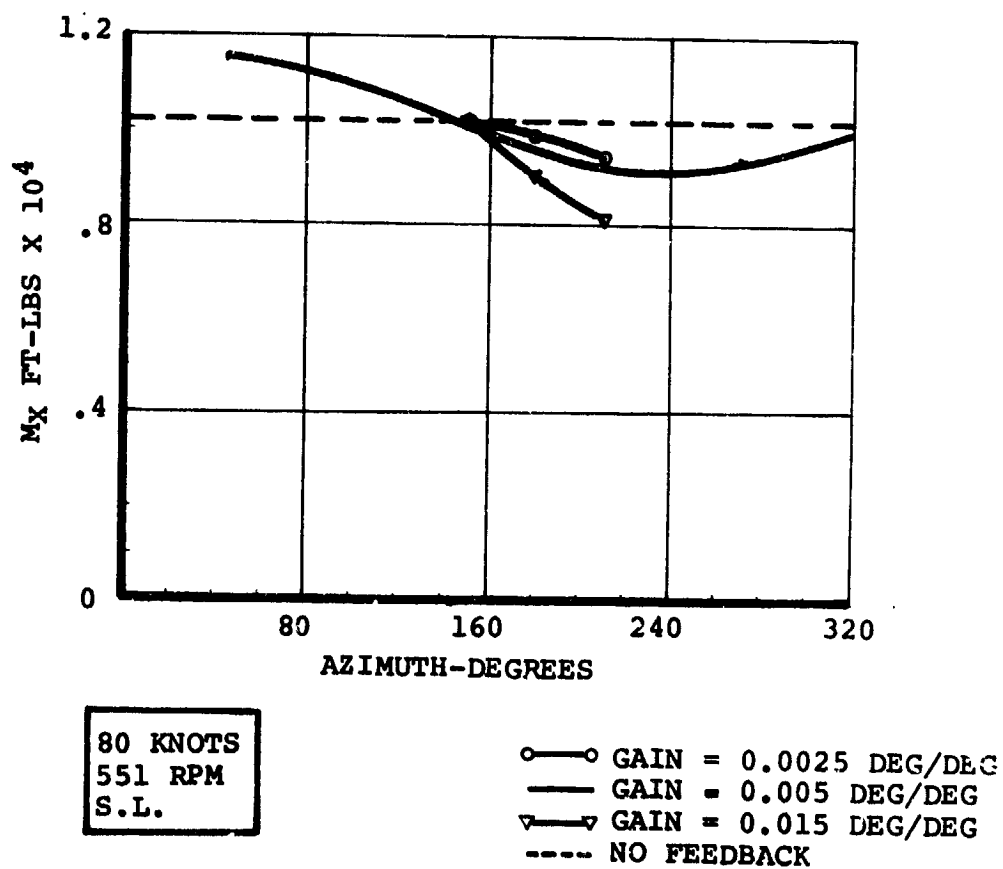
FIGURE 3.28. VARIATION OF HUB YAWING MOMENT WITH
AZIMUTH AND FEEDBACK GAIN AT 80 KNOTS
AND 551 RPM.
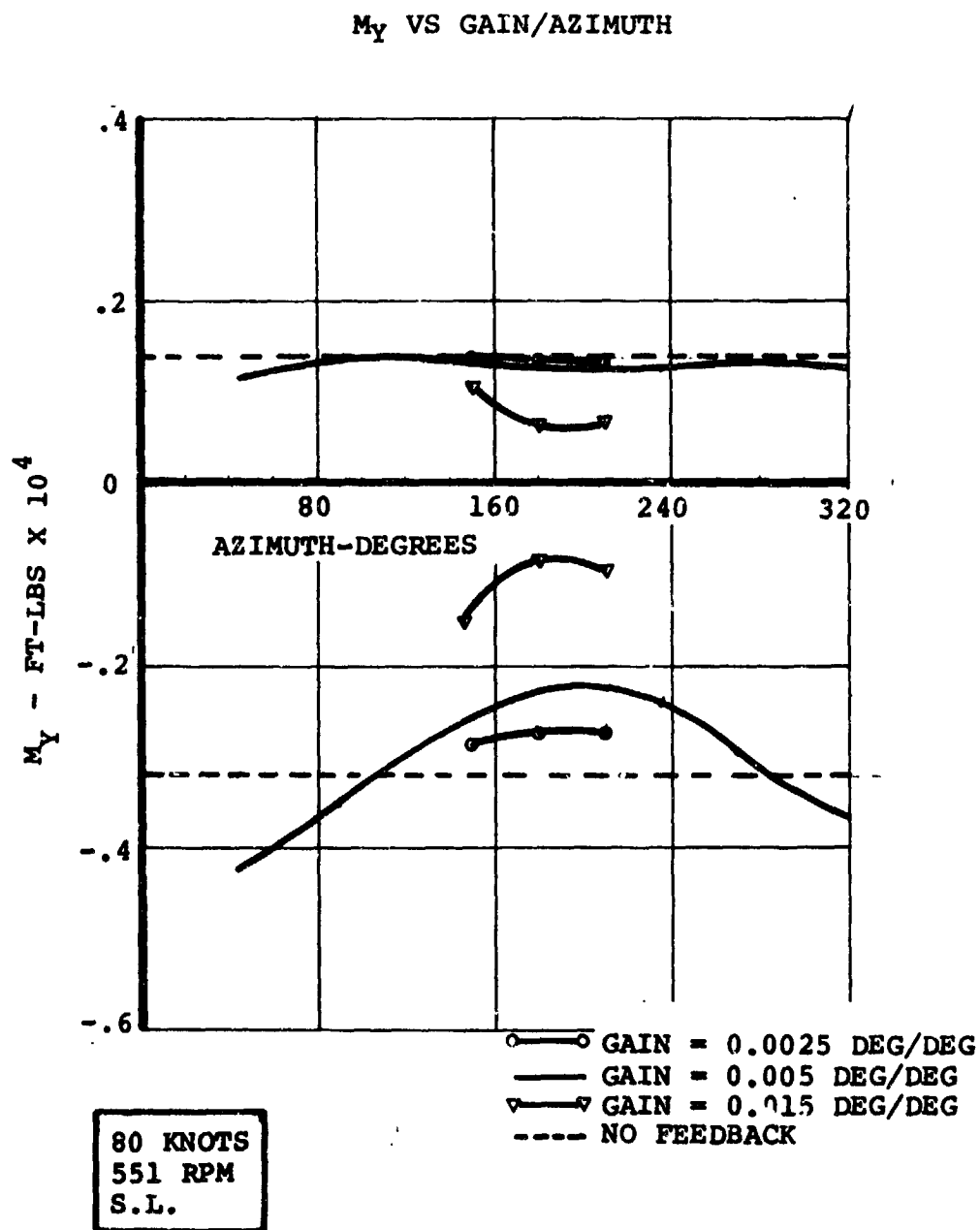
FIGURE 3.29. VARIATION OF HUB PITCHING MOMENT WITH AZIMUTH AND FEEDBACK GAIN AT 80 KNOTS AND 551 RPM.
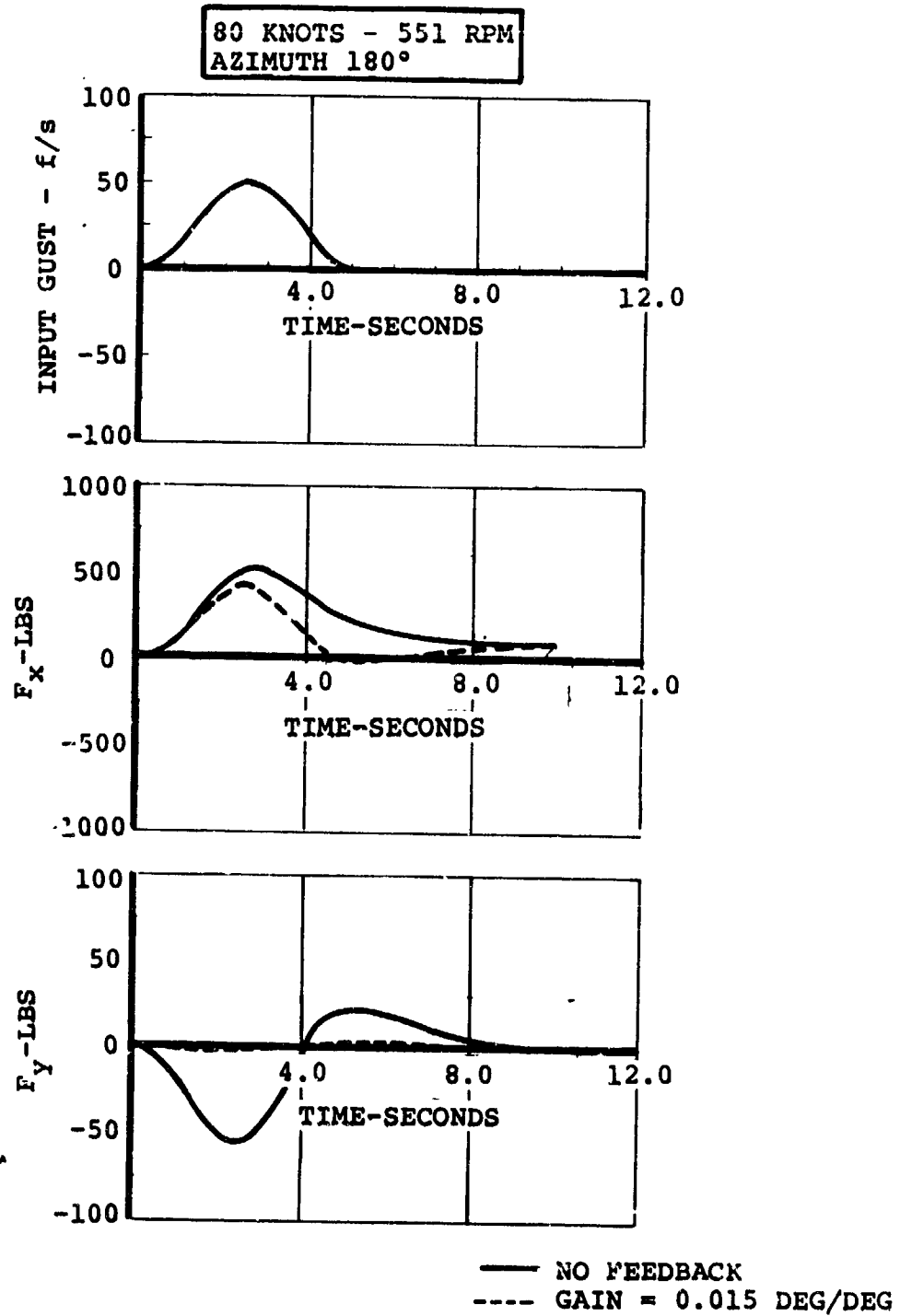
FIGURE 3.30. EFFECT OF FEEDBACK ON THE DYNAMIC RESPONSE
OF ROTOR HUB FORCES AT 80 KNOTS AND 551 RPM.

D222-10060-3
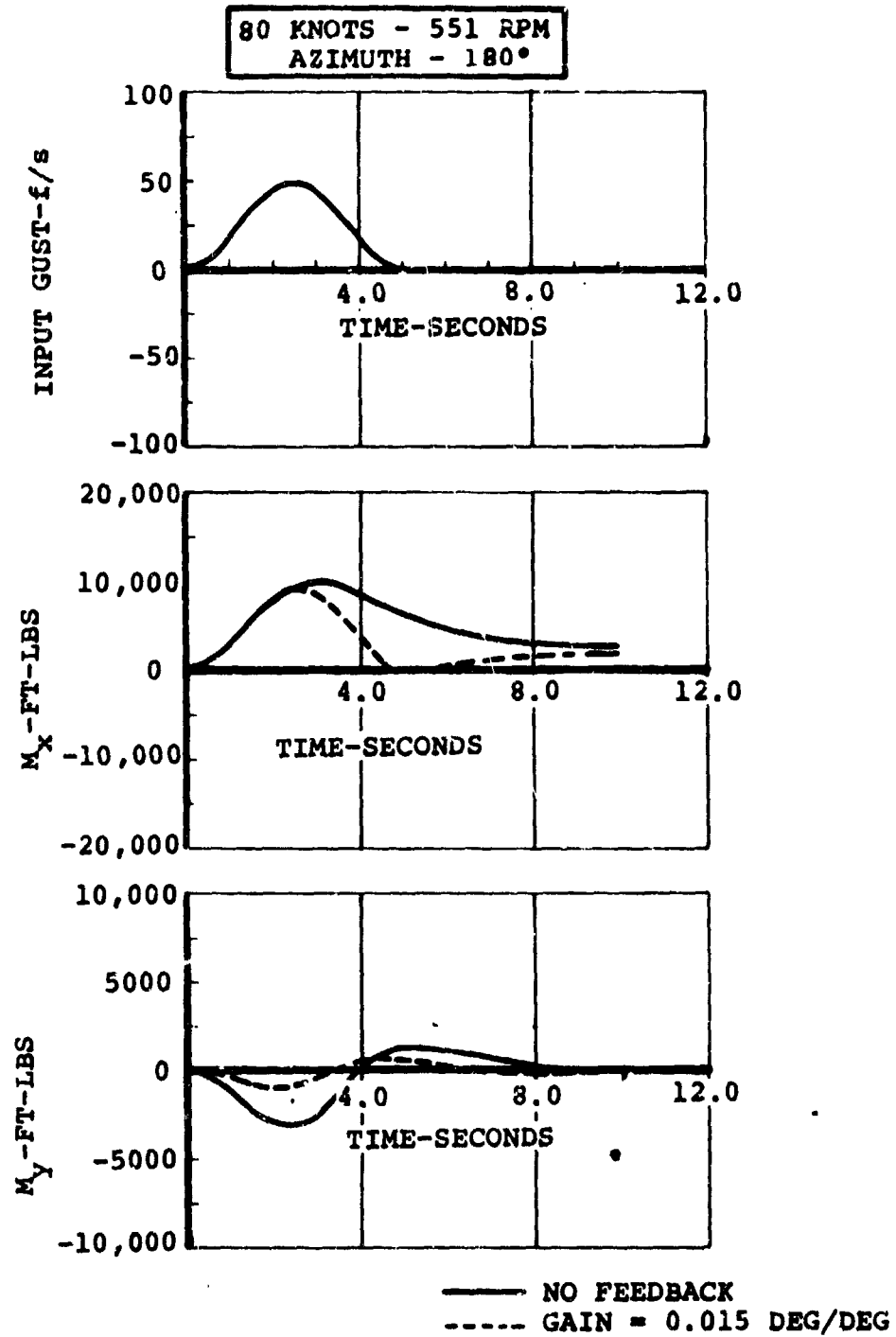
FIGURE 3.31. EFFECT OF FEEDBACK ON THE DYNAMIC
RESPONSE OF ROTOR HUB MOMENTS AT
80 KNOTS AND 551 RPM.

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D222-10060-3

FREQUENCY VS GAIN/AZIMUTH

80 KNOTS
551 RPM
S.L.

○────○ GAIN = 0.0025 DEG/DEG
──── GAIN = 0.005 DEG/DEG
▽────▽ GAIN = 0.015 DEG/DEG
──── NO FEEDBACK

FIGURE 3.32.   VARIATION OF SHORT PERIOD MODE FREQUENCY
WITH AZIMUTH AND FEEDBACK GAIN.
V=80 KNOTS.   Ω=551 RPM.

89

% CRITICAL DAMPING VS GAIN/AZIMUTH

FIGURE 3.33.   VARIATION OF SHORT PERIOD MODE DAMPING
WITH AZIMUTH AND FEEDBACK GAIN.
V=80 KNOTS.   Ω=551 RPM.

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one or more of a possible set of objectives. For optimal performance scheduling of feedback gain and azimuth as a function of flight condition, (speed and RPM) is necessary. Although the system demonstrated for 250 knots was even more effective at 350 knots, reduced speeds and increased RPM's call for different azimuth settings and reduced gain.

The response characteristics of the blade loads in dynamic situations may also be substantially improved by feedback. Differences in behavior noted between transition and moderately high speed cruise are caused by changes in the short period mode of the aircraft, and in the damping of the blade flapping mode. At 80 knots the short period mode is almost critically damped and its low frequency makes it unresponsive to the gust wave length assumed, with the result that blade loads show little tendency to overswing.

A promising area for additional analytically and experimental study will be the use of spoiler, flap and elevator feedback to modify the basic aircraft response characteristics so that the demands on a swashplate feedback load alleviation system are more nearly constant over the operating range.

The current study was performed for two conditions; the alleviation of loads due to a steady state condition and of those induced by flying through a discrete 5 second period gust. Other gust wavelengths must also be considered in

the design of a useful system, and this raises the question of compatibility from the standpoint of frequency. The nature of the blade load response varies with the type and frequency (period) of the loading condition.

Thus, variation of gain and azimuth with frequency is anticipated as a requirement. This may be accomplished using appropriate transfer functions to shape the signals in the A_1 and B_1 channels. At a given frequency the azimuth of the net feedback signal may be controlled by the relative frequency response in the two channels.

Relative phasing of the A_1 and B_1 signals may also be of value when unsteady loading conditions are encountered. The value of phasing the feedback signals applied to the swashplate is discussed in detail in Section 5. The investigation of such effects including the complimentary use of feedback to the conventional controls is recommended for further analytical and wind tunnel study. The general outline of further investigation and implementation is given in Section 9.

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4.   DIRECT LIFT FEEDBACK CONTROL

4.1   BACKGROUND AND OBJECTIVES

Alleviation of aircraft normal acceleration response to gusts has long been the objective of many aircraft design teams. With gust alleviation it was hoped to relieve structural loads, improve the ride qualities, lessen pilot fatigue, and/or provide a more stable flying platform. While many early attempts at gust alleviation were disappointing (References 4.1 4.2, & 4.3) it is thought that in many of these the attempt was being made without a full understanding of all aspects of flight dynamics and control system theory. Advances in avionics and in analytical and simulation techniques would today provide a much higher probability of success. In support of this view, attention is drawn to the striking success of systems with very similar objectives such as the LAMS system which is in operation on the B-52 bomber. This leads to the feeling that it is worthwhile to renew investigations of direct lift control systems using current feedback system technology and spoiler flap technology. The application of direct lift techniques in Tilt Rotor aircraft is particularly interesting for several reasons. One is that these aircraft will spend the greater portion of their time at low altitudes and thus be in turbulent air, and a system which responds to reduce the characteristically nigh lift curve slope of such aircraft would provide

distinct improvements in ride quality. Another consideration
is a high gust response will tend to limit the effectiveness of
the swashplate feedback blade load alleviation system. Since
blade response is stimulated by linear and angular accelerations
of the rotor hub and cyclic pitch control will not be particularly
effective in compensating for such inputs, direct lift feedback
may improve the effectivenes of the blade load alleviation
system in minimizing blade loads. Finally, in a typical tilt
rotor design such as the Model 222, provision of a direct lift
control system represents a relatively modest increment in de-
sign effort and hardware modification. Full span flaps and
spoilers are current features; automatic electrical control of
the full span flaps and symmetrical spoiler control are seen as
relatively minor modifications, along with automatic control
of the elevators to compensate for flap induced pitching moment.

Objectives of the gust alleviation system will be:

    O 40% to 60% reduction in normal acceleration
        response to gusts

    O no adverse effect on aircraft flying qualities

    O simplicity of system design

Evaluation of the gust alleviation system was made in three
parts. The first section discusses the approach to alleviating
gusts, the math model, and the feedback loops. The second
section discusses the results in terms of gain, system stabi-
lity and gust alleviation. In the third section some

conclusions are drawn about the effectiveness of the system and its practicality for a tilt rotor aircraft.

4.2   GUST ALLEVIATION SYSTEM APPROACH

A tilt rotor aircraft design with full span spoilers and flaps has been used to evaluate a gust alleviation system using direct lift control (DLC). For gusts which tend to increase the load factor, spoilers are deflected to "dump" wing lift and thus negate the effect of the gust. For gusts which decrease load factor, flaps are deflected thus increasing lift and negating the gust response. Both spoiler and flap deflection generate an aircraft pitching moment which could be trimmed by gearing the elevator to both the spoiler and the flaps. A schematic of the system is shown in Figure 4.1. First order approximations for f' p, spoiler, and elevator actuators have been made. In each case τ_{ACT} = .05 seconds.

To keep the system as simple as possible no filtering has been employed and only a minimum of signal shaping to indicate whether the flap or spoiler should be deflected was used. Aircraft angle of attack was sensed rather than normal acceleration in order to achieve some lead on aircraft response. The time lead is of two forms. First, if angle of attack is sensed at the nose of the aircraft there will be a finite time until the gust reaches the wing. The time lead increment will be of the form

$$\Delta t_1 = \frac{\text{(length from nose to wing leading edge)}}{\text{(aircraft velocity)}}$$

95
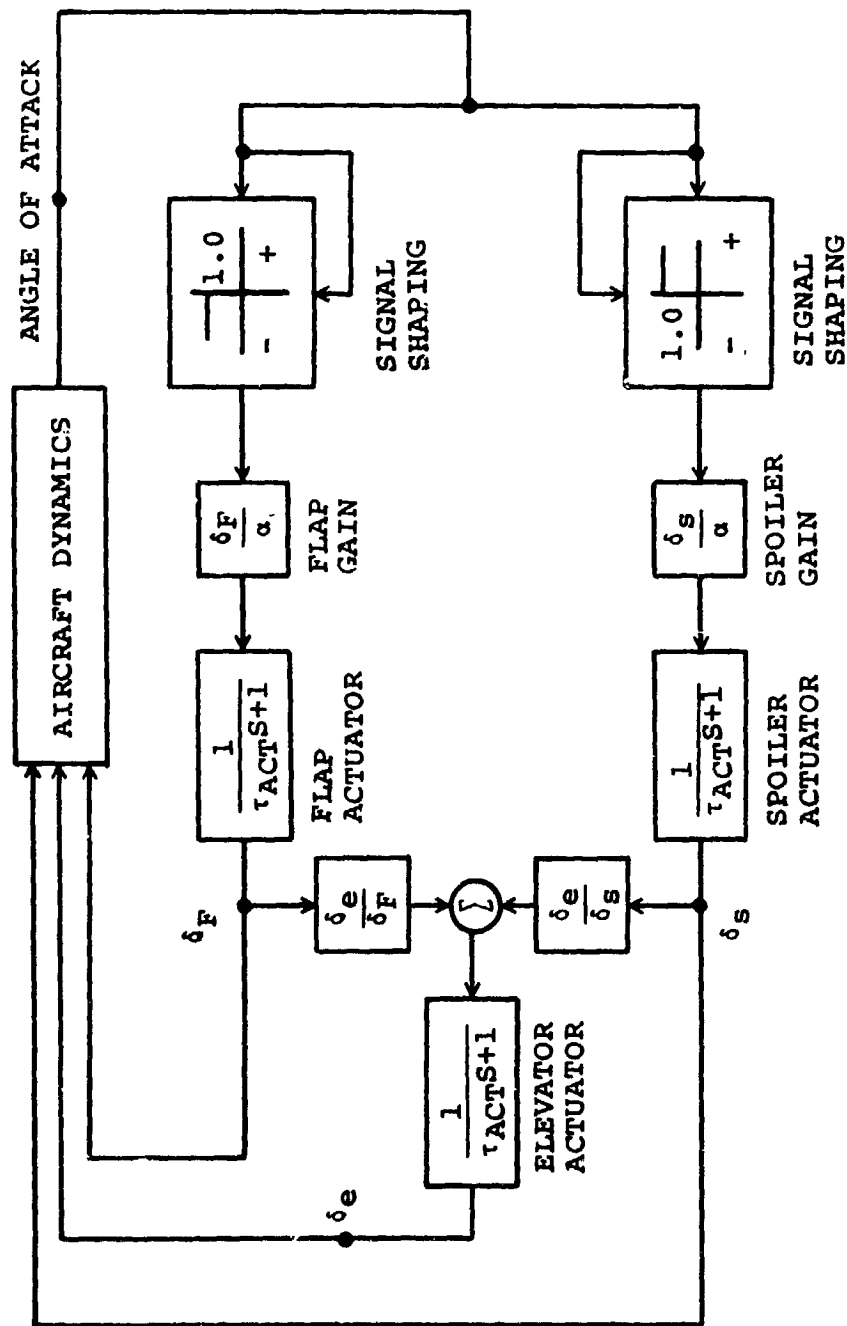
FIGURE 4.1. SCHEMATIC OF DIRECT LIFT CONTROL
GUST ..LLEVIATION SYSTEM

FIGURE 4.2. EFFECT OF GUST ALLEVIATION SYSTEM GAIN
ON NORMAL ACCELERATION RESPONSE
250 KNOTS

NO FEEDBACK $\rightarrow$ $\leftarrow$ $\overline{1.}$ SECOND

$\delta_F/_\alpha = 2.47\%$ $\delta_S/_\alpha = 4.87\%$

$\delta_F/_\alpha = 4.95\%$ $\delta_S/_\alpha = 9.75\%$

$\delta_F/_\alpha = 7.43\%$ $\delta_S/_\alpha = 14.61\%$
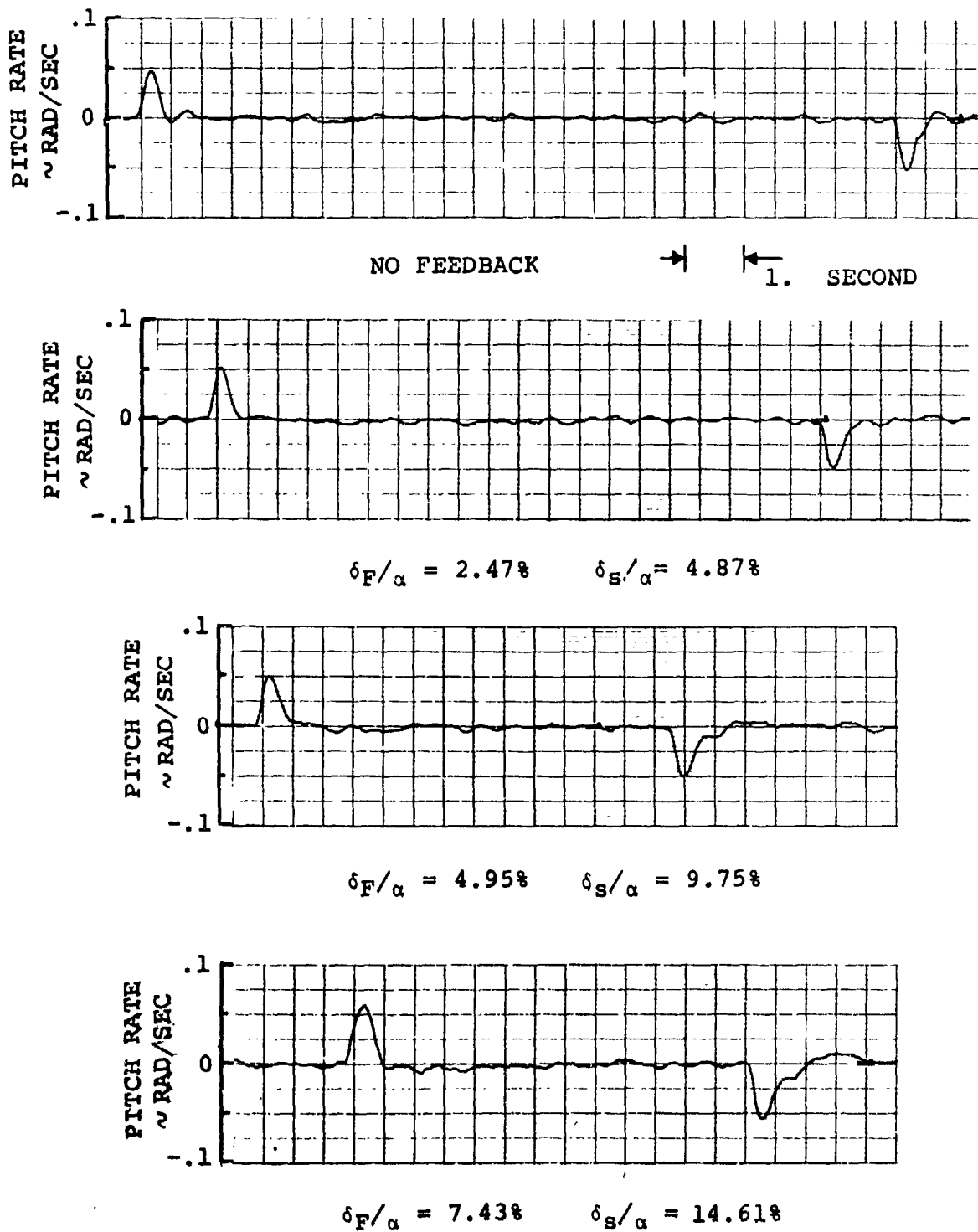
FIGURE 4.3. AIRCRAFT PITCH RATE RESPONSE TO RAMP GUST
WITH AND WITHOUT DLC GUST ALLEVIATION SYSTEM

and 4.4 show the normal acceleration and pitch rate time history responses for various spoiler and flap gains. In this configuration the pitching moment due to small deflection of spoiler and flap is of such magnitude that elevator was not deemed necessary to trim the moments. The gain level can have a significant effect on aircraft flying qualities. At higher levels of gain the short period mode seems to be adversely affected as seen by the traces of Figures 4.3 and 4.4. A more detailed study of the system dynamics will have to be performed to determine if this problem can be overcome so that further effectiveness can be accomplished. In the second portion of the 250 knot study the aircraft was flown through random turbulence with the system off and the system on (Figures 4.5 to 4.8). The same gain variation as with the ramp gust was applied. At a gain level of $\delta_F/\alpha = 4.95$ and $\delta_S/\alpha = 9.75$, a 45% reduction in normal acceleration gust response was achieved. Although this level of gust alleviation is not as great as for the ramp gust, it is significant and indicates that with further study of possible signal shaping a gust alleviation level of 60% could be reached and possibly more.

At a velocity of 100 knots and a nacelle incidence of 25-degrees, the same DLC gust alleviation system was evaluated. The aircraft was flown through random turbulence and the actual and RMS normal acceleration responses were measured. Aircraft normal acceleration gust response was reduced by 15% in the
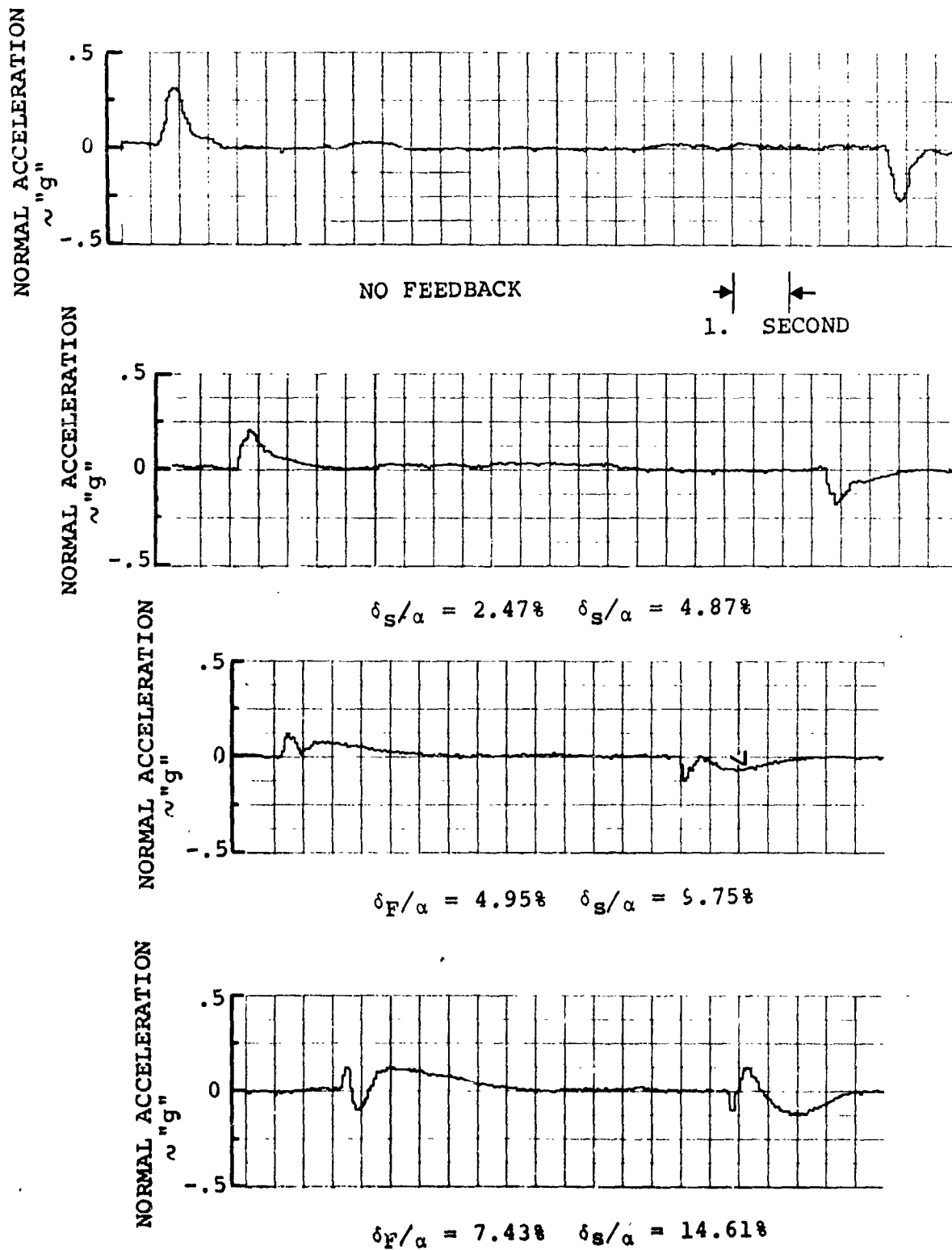
FIGURE 4.4. AIRCRAFT CABIN NORMAL ACCELERATION RESPONSE
TO RAMP GUST WITH AND WITHOUT DLC GUST
ALLEVIATION SYSTEM
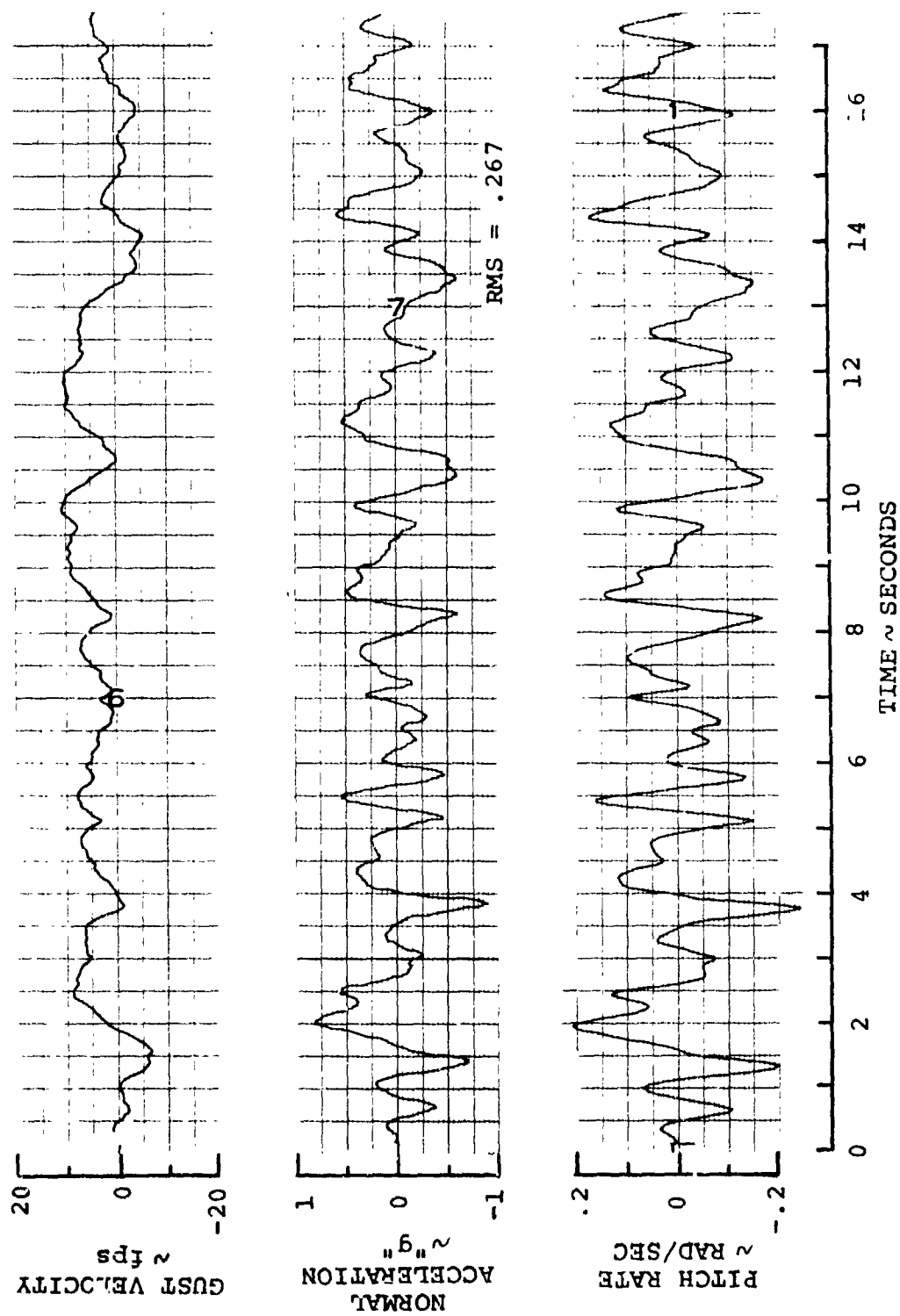
FIGURE 4.5. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
250 KNOTS   NO FEEDBACK
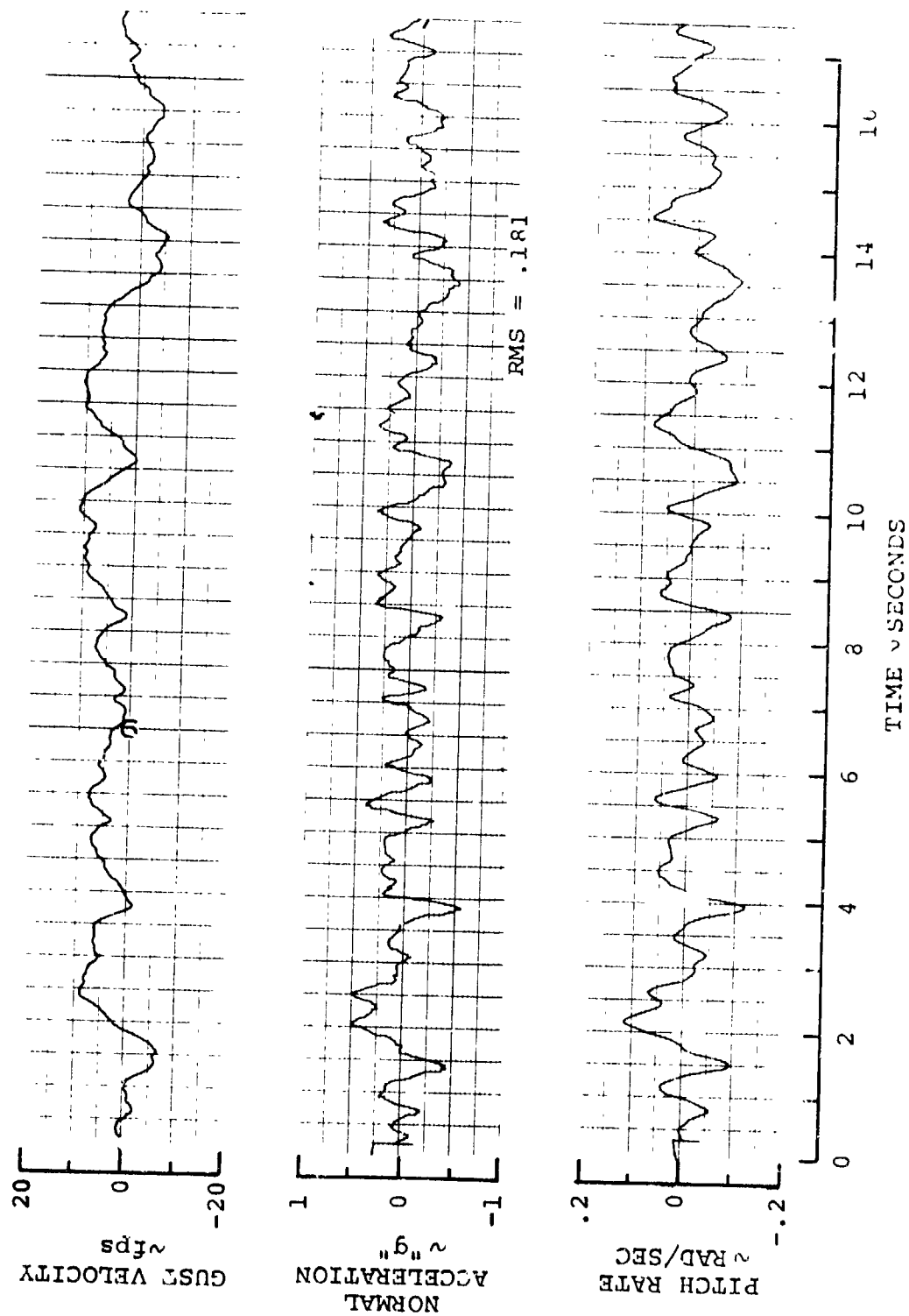
FIGURE 4.6. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
250 KNOTS $\delta_F/_\alpha$ = 2.47 $\delta_S/_\alpha$ = 4.78
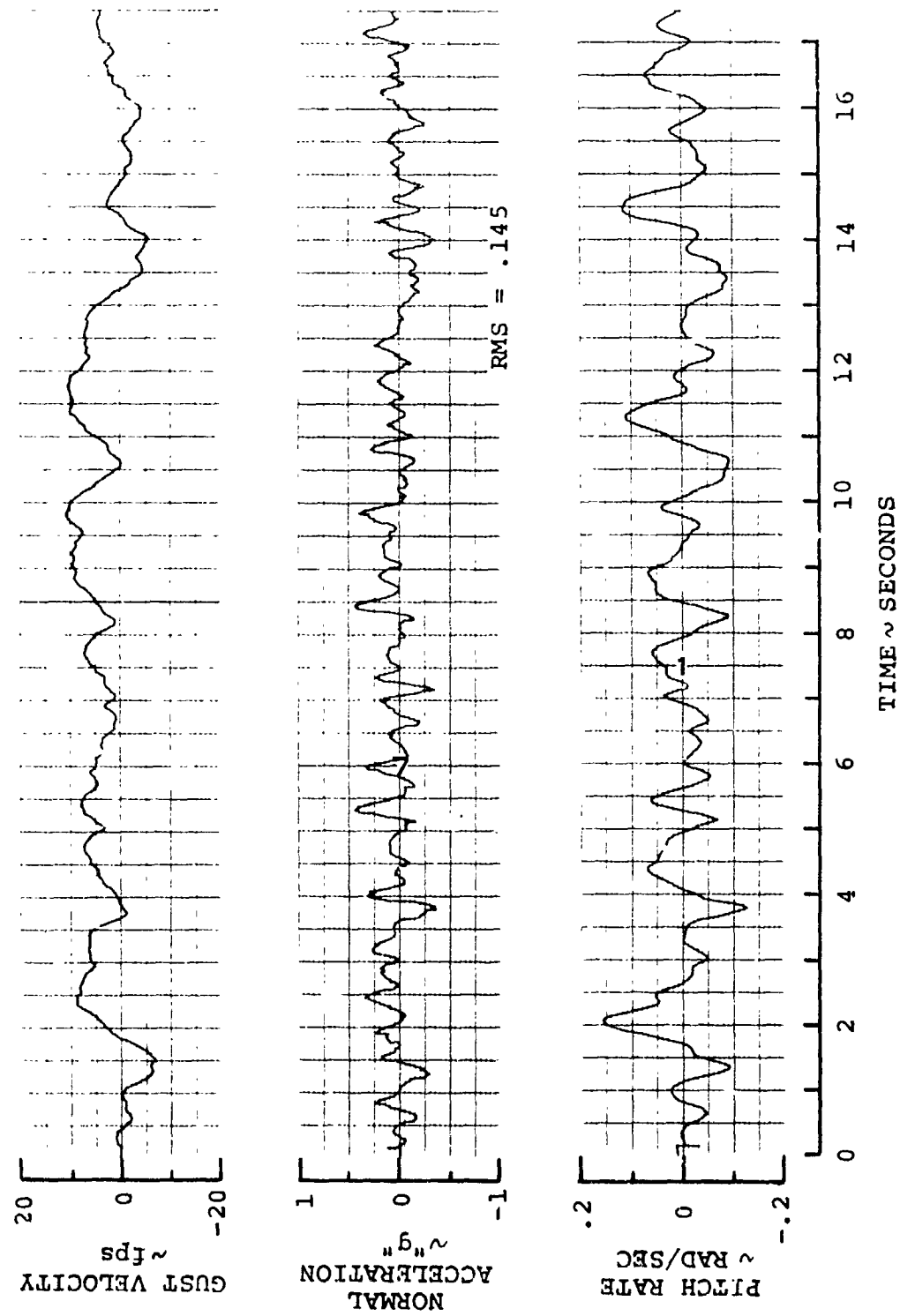
FIGURE 4.7. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
250 KNOTS $\delta_F/_\alpha = 4.95$ $\delta_S/_\alpha = 9.75$
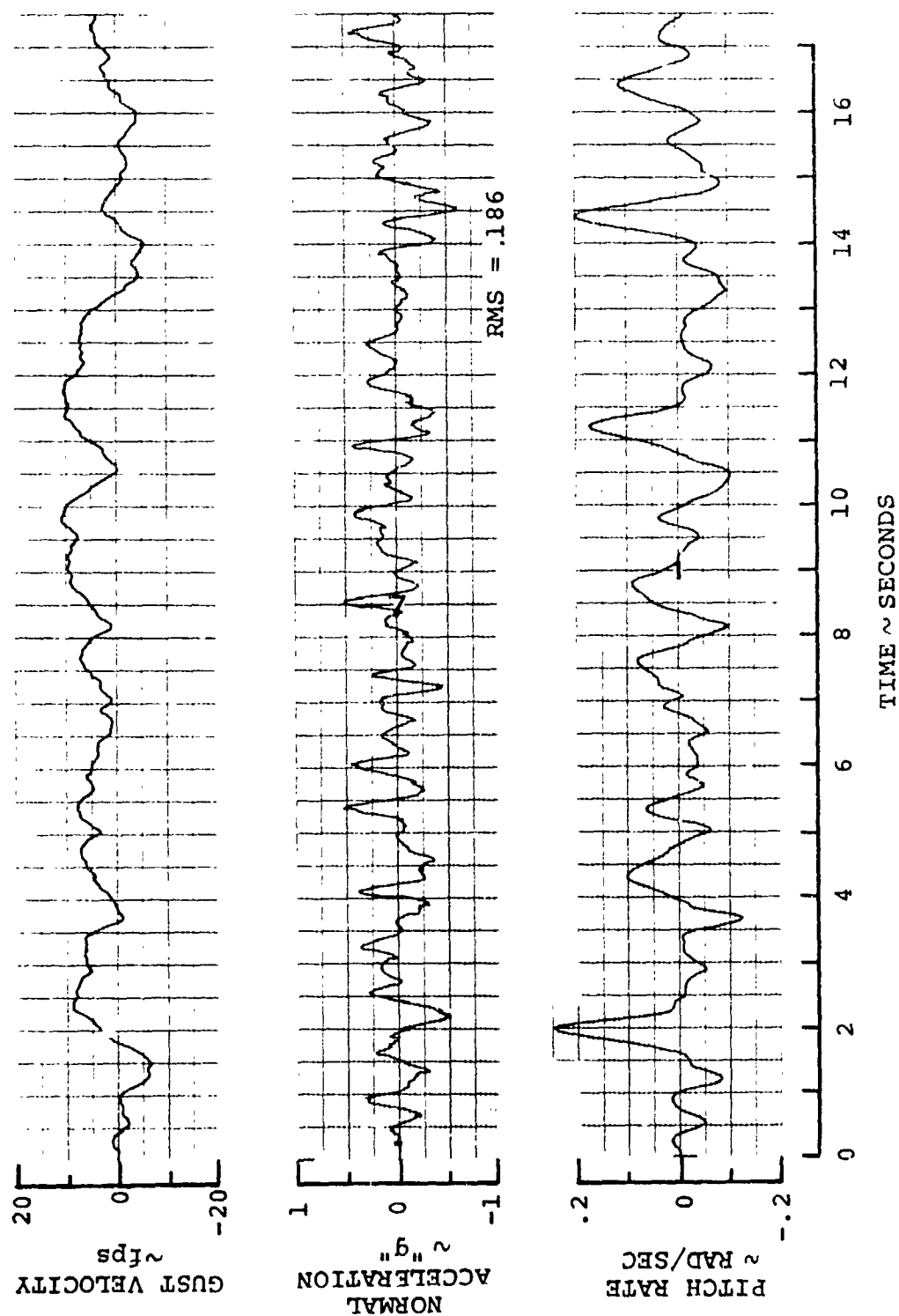
FIGURE 4.8. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
250 KNOTS δ_F/α = 7.43 δ_S/α= 14.61

best configuration. As seen in Figures 4.9 through 4.13 the
aircraft dynamic response seems to be more affected than at
250 knots. It is evident that at certain values of gain the
phugoid mode is excited in conjunction with the short period
mode - resulting in an increase in RMS response level rather
than a decrease. At 100 knots the dynamic responses of the
aircraft are more complicated because of the rotor shaft
angle and the slower response of the aircraft at this
velocity. A more detailed design of the DLC gust alleviation
system and further analytical studies are required to ensure
that other aircraft modal responses are not degraded. Thus,
flap and spoiler operation may actually excite the wing
bending mode, so that an associated modal suppression system
as discussed in Section 6 might be needed to counteract this
tendency. This should be the subject of further work.

Throughout the evaluation of this system, particular concern
was paid to spoiler and flap deflection rates. At all gain
variations the maximum spoiler rate was 120-degrees per second
and the maximum flap rate was 80-degrees per second. Minimum
design   rates for these surfaces for roll control are 110-
degrees per second and 50-degrees per second respectively, so
that the rates required for gust alleviation represent only
modest increases.

DLC gust alleviation using an angle of attack sensor has been
shown to be of sufficient merit to warrant further investigation.

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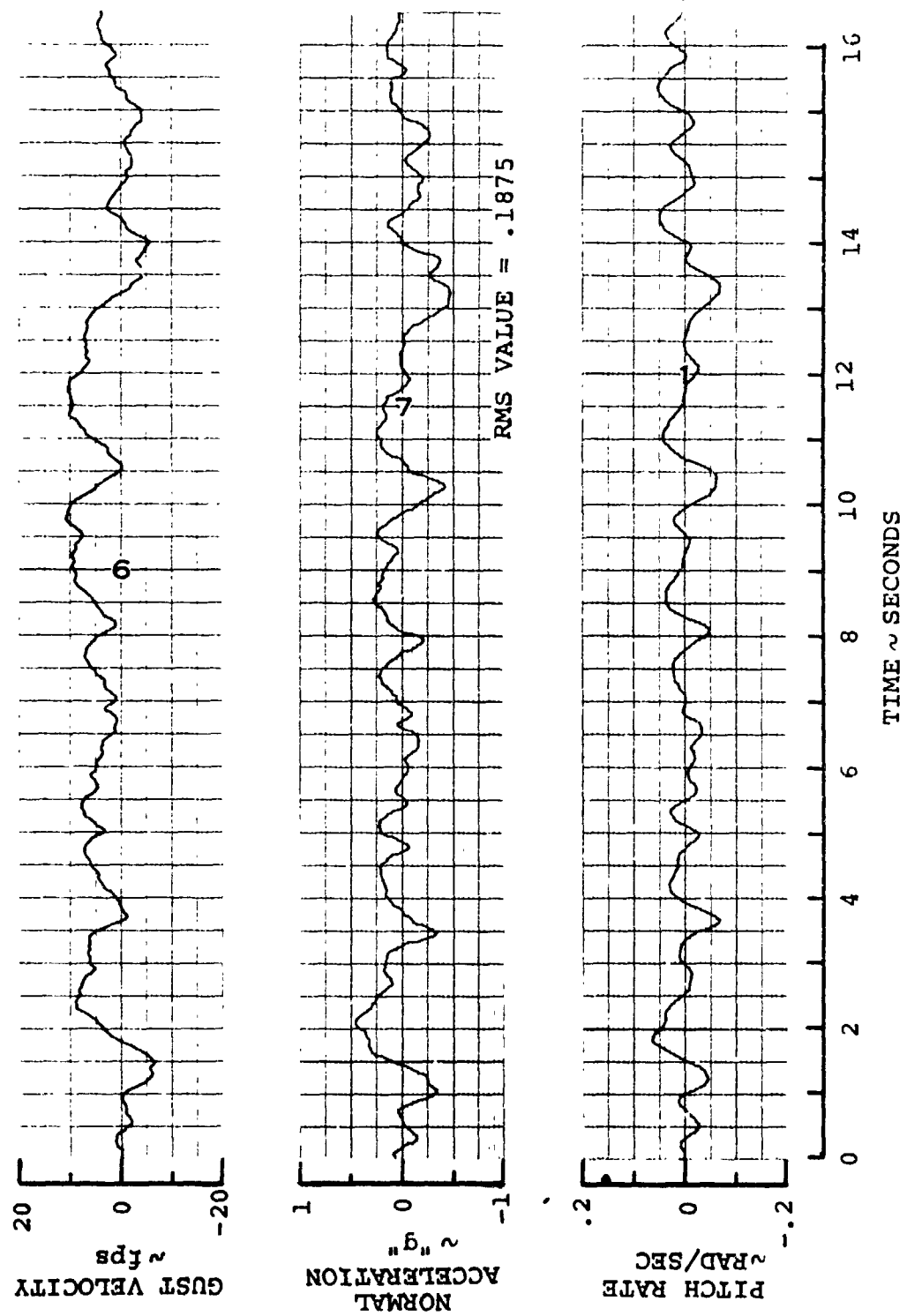
FIGURE 4.9. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
100 KNOTS   NO FEEDBACK
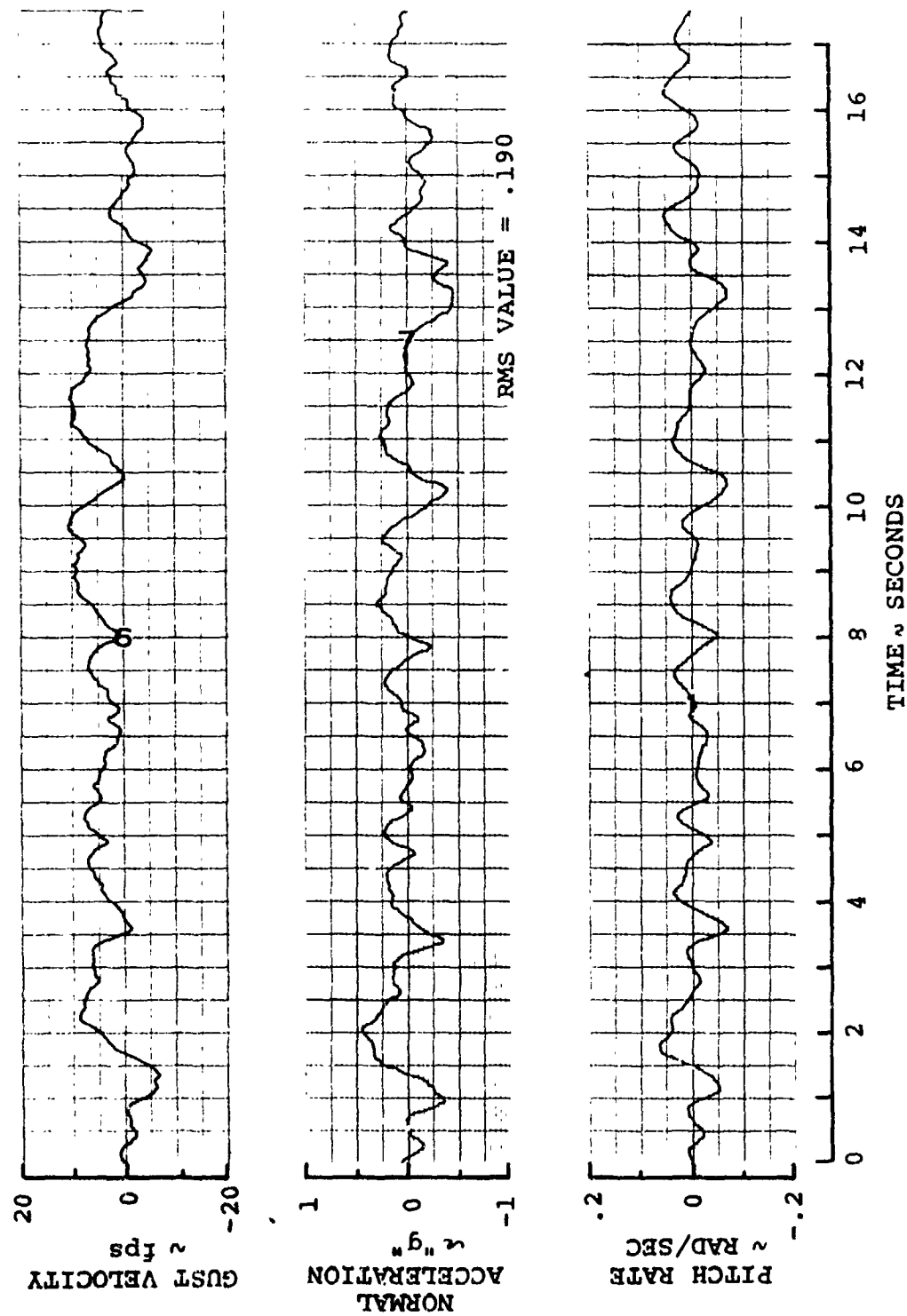
FIGURE 4.10. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
100 KNOTS $-\delta_F/_\alpha = 1.24 - \delta_S/_\alpha = 2.44$
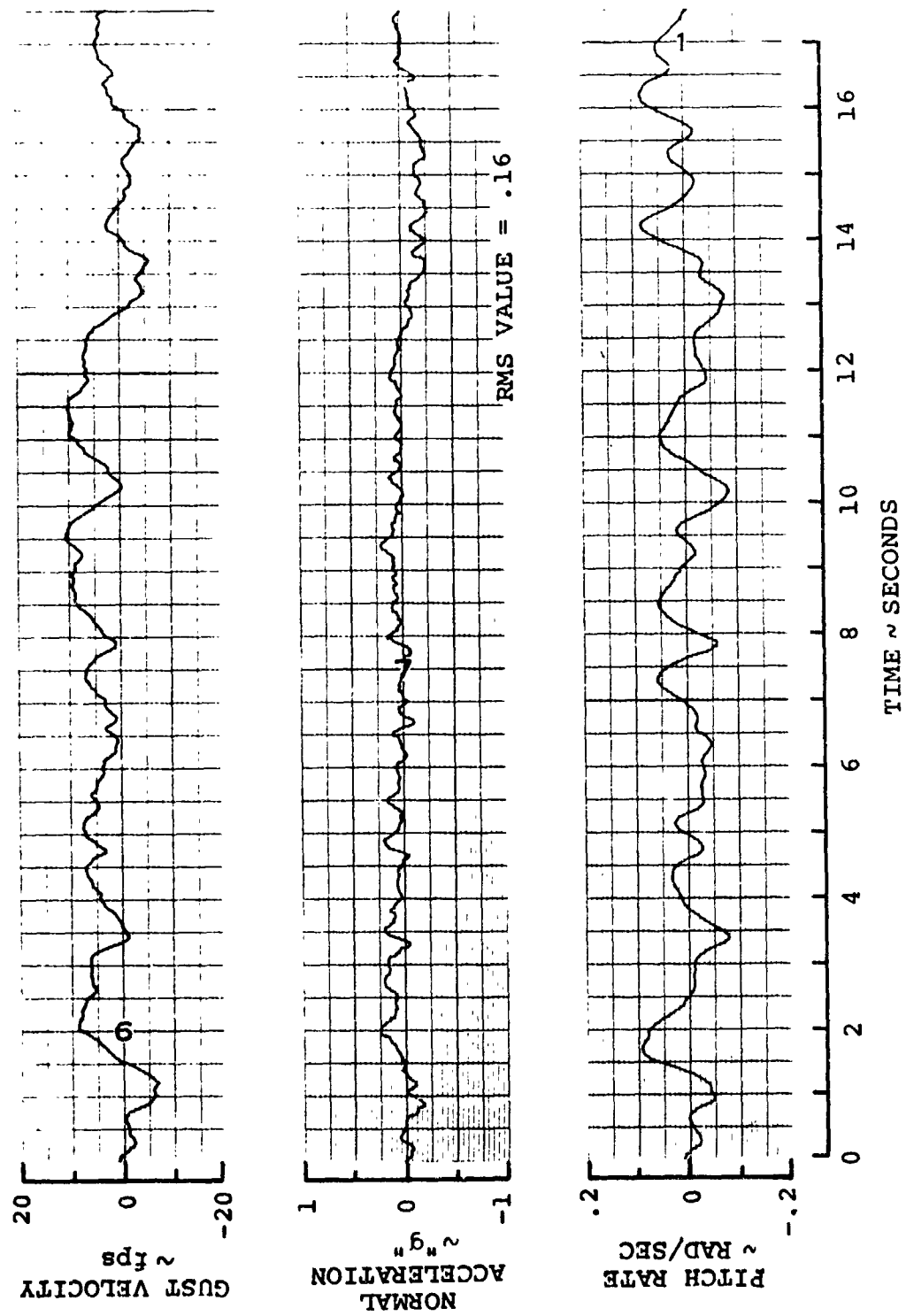
FIGURE 4.11. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
100 KNOTS $\delta_F/_\alpha = 2.47$ $\delta_S/_\alpha = 4.78$
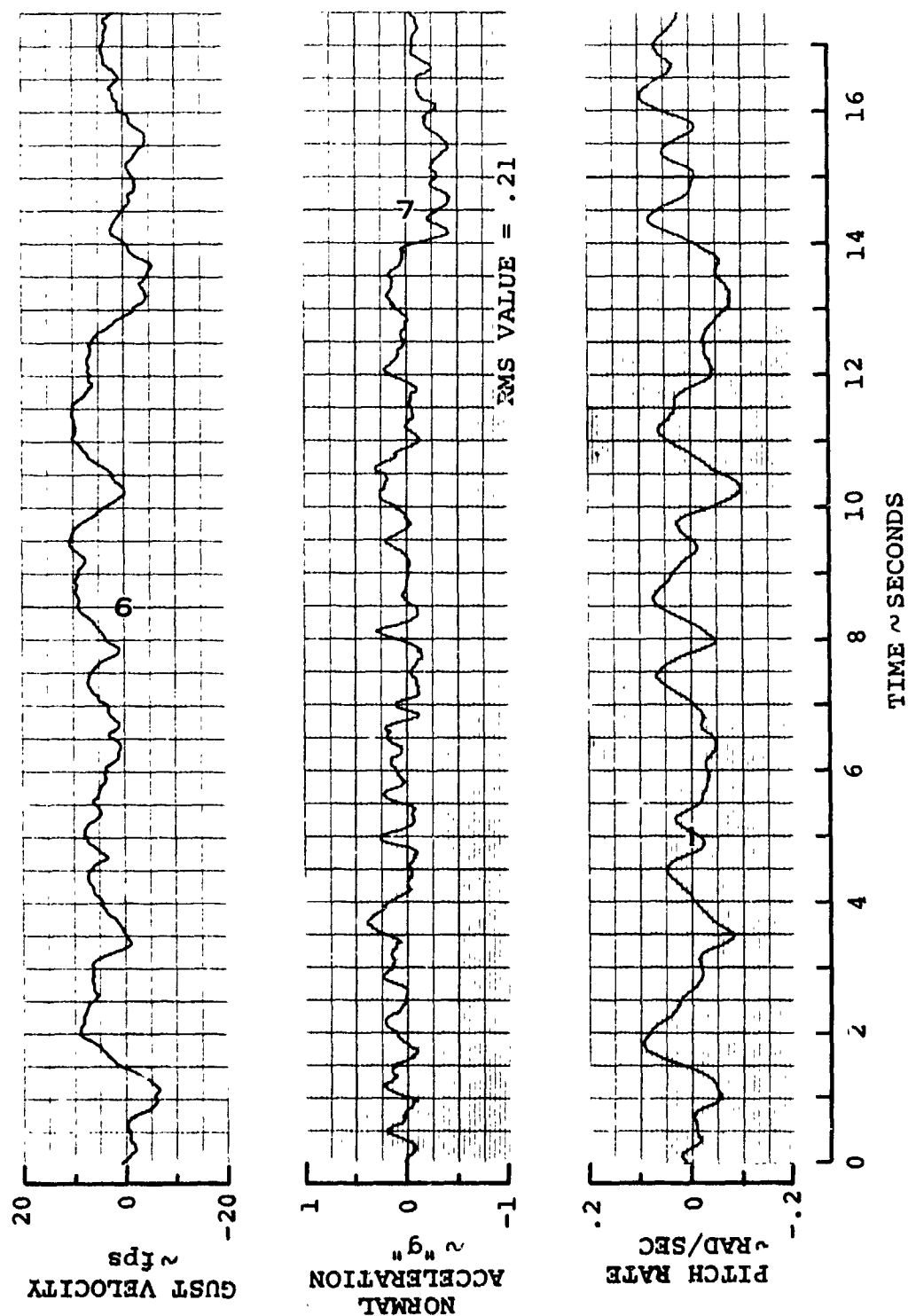
FIGURE 4.12. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
100 KNOTS $\delta_F/\alpha = 4.95$ $\delta_{\Bbb{B}'}/\alpha = 9.75$
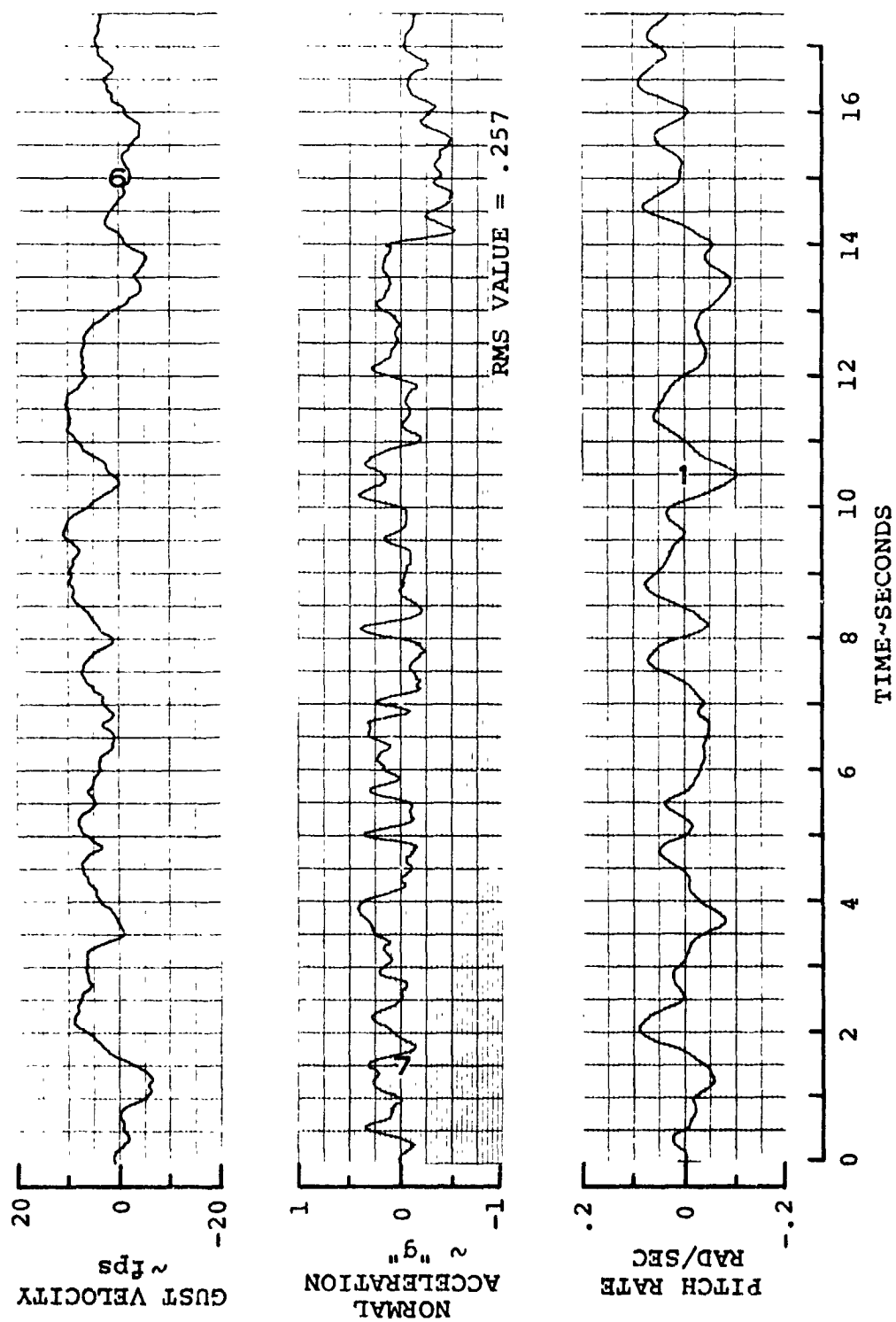
FIGURE 4.13. AIRCRAFT RESPONSE TO RANDOM TURBULENCE
100 KNOTS $\delta_F/_\alpha = 7.43$ $\delta_S/_\alpha = 14.61$

At 250 knots a 45% alleviation of the normal acceleration
response to random turbulence was shown and at 100 knots 15%
alleviation was seen.

4.4 CONCLUSIONS

This study has proven the feasibility of DLC gust alleviation
in the cruise flight regime and has shown that such a system
has promise in the transition, approach and landing regime.
In the 250 knot cruise configuration a 45% reduction in the
response to random turbulence was noted. At optimum gains the
aircraft short period response was not adversely affected. In
a 100 knot transition configuration with nacelle incidence of
25-degrees a 15% reduction in normal acceleration response
was se   Sc   ⁻.verse affect on aircraft flying qualities
— - ⸌   . but it is believed that with further study and the
⸗ of elevator feedback these problems may be overcome.

Mechanization of the system on a full scale aircraft presents
no major problems. Spoiler and flap actuators which can pro-
duce rates of 120-degrees per second and 80-degrees per second
respectively will be required for the task. Some additional
study is required to determine an optimum configuration and
location for the angle of attack sensor.

5.   MODAL SUPPRESSION AND AEROELASTIC STABILITY AUGMENTATION

5.1 BACKGROUND AND OBJECTIVES

The concept of using swashplate feedback for modal suppression
and stability augmentation in tilt rotor aircraft is attractive
for a number of reasons wich are listed below.

O Benefits: 1. Comfort. The mass distribution of a typical
   tilt rotor aircraft such as the M222 (i.e., large mass con-
   centration at the wing tips) is such that a responsive or
   lightly damped wing mode may result in an uncomfortable
   level of cabin vibration at the wing frequency. Cabin acce-
   lerations will be of the order of 50% of wing tip accelera-
   tions in the fundamental modes at frequencies near the body
   frequencies of the crew. 2. Safety. While the aircraft
   will be built, under current design philosophy, with adequate
   margins of stability, the presence of a correctly designed
   modal suppression system would increase margins to the point
   where loss of wing stiffness due to partial structural fail-
   ures would not be catastrophic. 3. Advanced Configurations.
   Development of successful and reliable modal suppression and
   aeroelastic stability augmentation systems would open up
   the path to more advanced design concepts. Currently wing
   design is constrained by stiffness requirements associated
   with the stability of wing/pylon/rotor dynamic interactions
   and this has associated penalties in weight and performance.
   Reduction of stiffness constraints would permit the use of
   a thinner wing with no weight penalty.

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O Cost: Since SAC swashplate actuators are already present
   for the SAS and blade load alleviation system, no major
   new hardware items are involved; only sensors, which are
   envisioned as accelerometer and signal conditioning avionics,
   are required additionally.

O Effectiveness: Rotor forces applied at the wing tip are
   highly effective in forcing or damping the modes.

In the immediate future, however, the value of modal suppression
systems will lie in improvement in ride qualities in turbulent
air, and potential improvement in the fatigue life of the wing
and blades. Studies for the SST, and operational equipment for
the B-52 have shown that feedback (modal suppression devices)
improve the structural gust response and the aircraft ride
qualities. Studies for modal suppression of the SST may be re-
viewed in Boeing Document D3-7600-10 and D3-7600-11.

Longer term applications would address battle damage survival
where the loss of wing stiffness due to partial destruction
would be compensated for by feedback.

In a new generation of aircraft reliance on feedback might be
proposed as the primary source of aeroelastic stability.

5.2 RATIONAL OF CURRENT STUDY

The study presented below concentrates on the use of swashplate
feedback to augment the damping in the wing vertical bending
mode characteristics of the Boeing Vertol M222 Tilt Rotor
Aircraft are used for the investigation. This

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mode was selected, (rather than the whirl flutter mode for example), because its response has a more direct association with near term applications. Also test data in relation to wing bending mode suppression was to be acquired and the analytical conclusions could be checked against test data.

For a tilt rotor aircraft with a soft in-plane rotor, the wing vertical bending mode has been shown to be important from several aspects:

- O gust response
- O aircraft weight
- O air resonance.

If response in the wing vertical bending mode could be altered or suppressed, improvement in the above categories could be made. Swashplate feedback has been proposed as a means for vertical bending modal suppression. Investigation of this feedback has proceeded in three distinct segments:

O initial investigations with a simplified system concept

O introduction of actual hardware characteristics and refined approach to optimization

O proposed use of a "wobbling" swashplate to obtain maximum effectiveness in suppressing the vertical bending mode.

Initial investigations were primarily exploratory in nature to determine what could be done to improve wing vertical bending response. As shown in Figure 5.1, a perfect feedback loop was envisioned. Filtering, actuator dynamics or sensor dynamics were not been considered. Results were encouraging
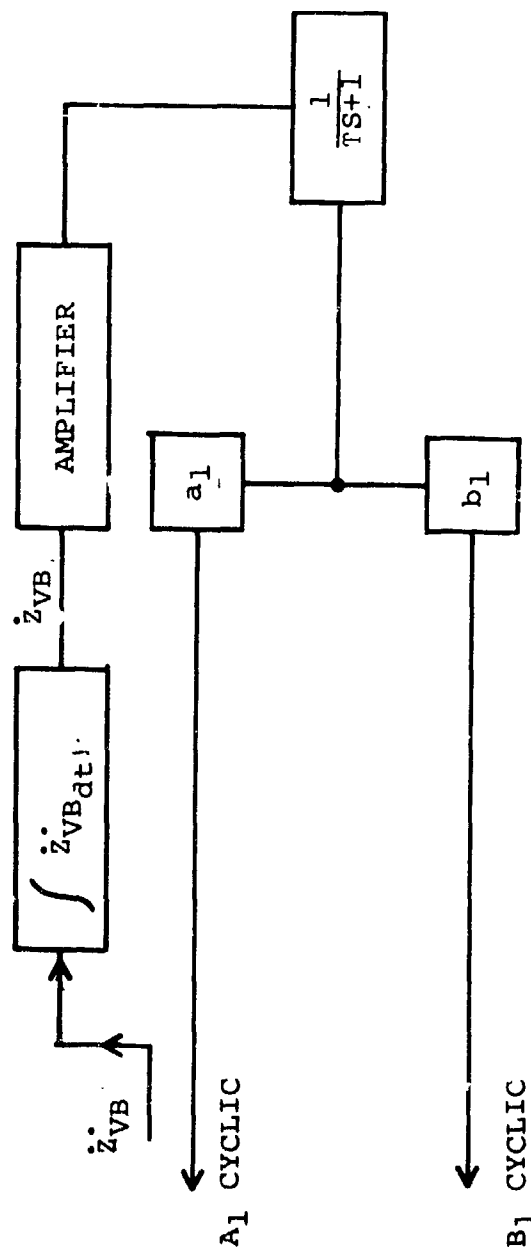
FIGURE 5.1. SCHEMATIC OF SIMPLIFIED AEROELASTIC STABILITY AUGMENTATION SYSTEM.

and lead the research to the second phase of study.

The second phase of the study used the system of Figure 5.2. This was felt to be fully representative of an actual hardware system. Representation of filters, actuators, sensors and phase shifters were made using actual frequency response calibrations. A_1 and B_1 feedback signals were assumed to be in phase. In this system a more refined approach was taken with special consideration to possible trouble spots. If trouble was located, "fixes" were proposed and evaluated. Four different styles of feedback were evaluated:

O   acceleration feedback with bandpass filter and phase shifter

O   acceleration feedback without filter and phase shifter

O   rate feedback

O   position feedback

The above systems were synthesized and analyzed using the combined tools of Bode analysis and root locus plots. The systems were evaluated with all the wing fundamental structural modes present, but not rigid body freedoms, since it was planned to measure the analytical predictions against data generated testing a cantilevered wing/rotor system in the NASA-Ames 40x80-foot wind tunnel.

In a third phase of the study a system using a "wobbling" swashplate and a possible research program is discussed. It is believed that

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FIGURE 5.2 SCHEMATIC OF AEROELASTIC STABILITY AUGMENTATION
SYSTEM FOR AIR RESONANCE MODE – WING BENDING

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this style system would make most useful effect of available cyclic pitch. With this viewpoint the advantages and dis-advantages of the system are specified.

5.3 MATHEMATICAL MODEL FOR MODAL SUPPRESSION STUDIES

A windmilling rotor semi-span cantilevered model was used as the basis for each of the studies. Included in the model were blade flap and lag degrees of freedom represented by coning and cyclic modes of the rotor. Fully coupled wing vertical bending, chord and torsion modes were included. Figures 5.3 and 5.4 define the frequency and damping spectrum for the model with the rotor spinning at 386 RPM. There were slight differences in the math model for the initial studies which reflected the differences between the 1/9 scale model and the 26-foot rotor full scale test.

There is a high degree of confidence that the mathematical model is representative of the physical system desired. Figures 5.5 through 5.10 show the correlation of the mathematical model analysis with actual test results. Note the high degree of correlation for the wing vertical bending mode. Attempts to excite the other modes were not very successful, as anticipated, which was one of the reasons for selecting the vertical bending mode for study.

The mathematical model of the feedback system provides six degrees of freedom which have the flexibility to represent nearly anything the analyst would desire in the way of transfer functions. A detailed description of the feedback capability
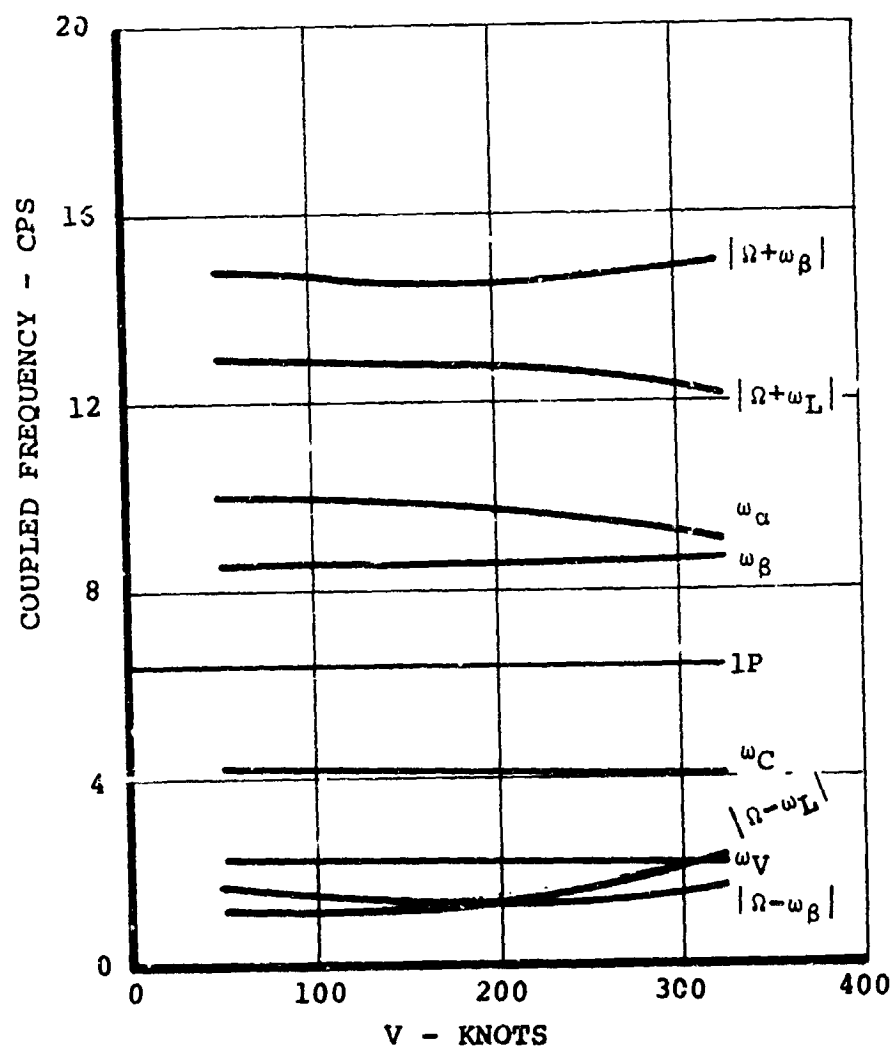
FIGURE 5.3. EFFECT OF VELOCITY ON TILT ROTOR
MODAL FREQUENCY

120
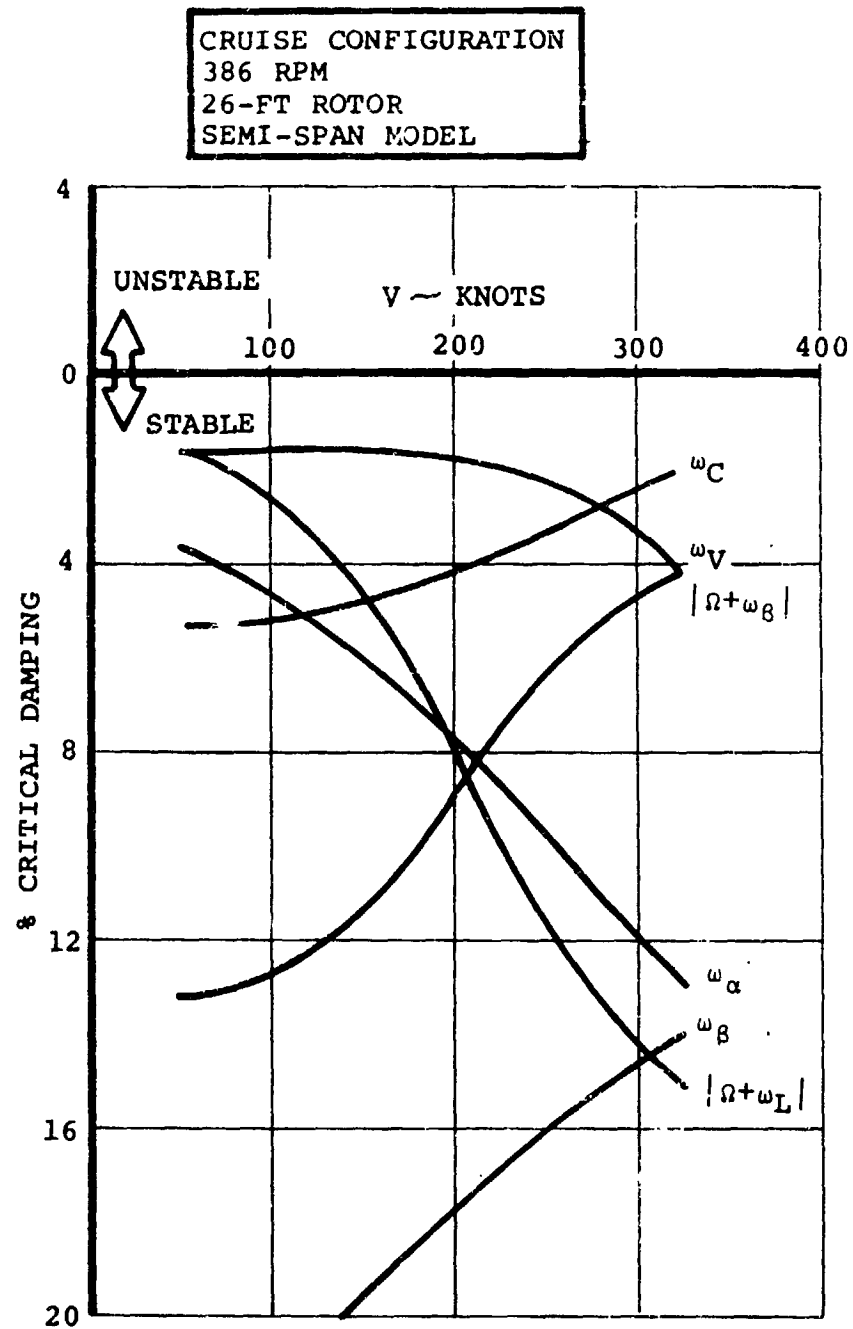
FIGURE 5.4. EFFECT OF VELOCITY ON TILT ROTOR
MODAL DAMPING.

NASA AMES 40 X 80 WIND TUNNEL
TEST 410

100 KNOTS
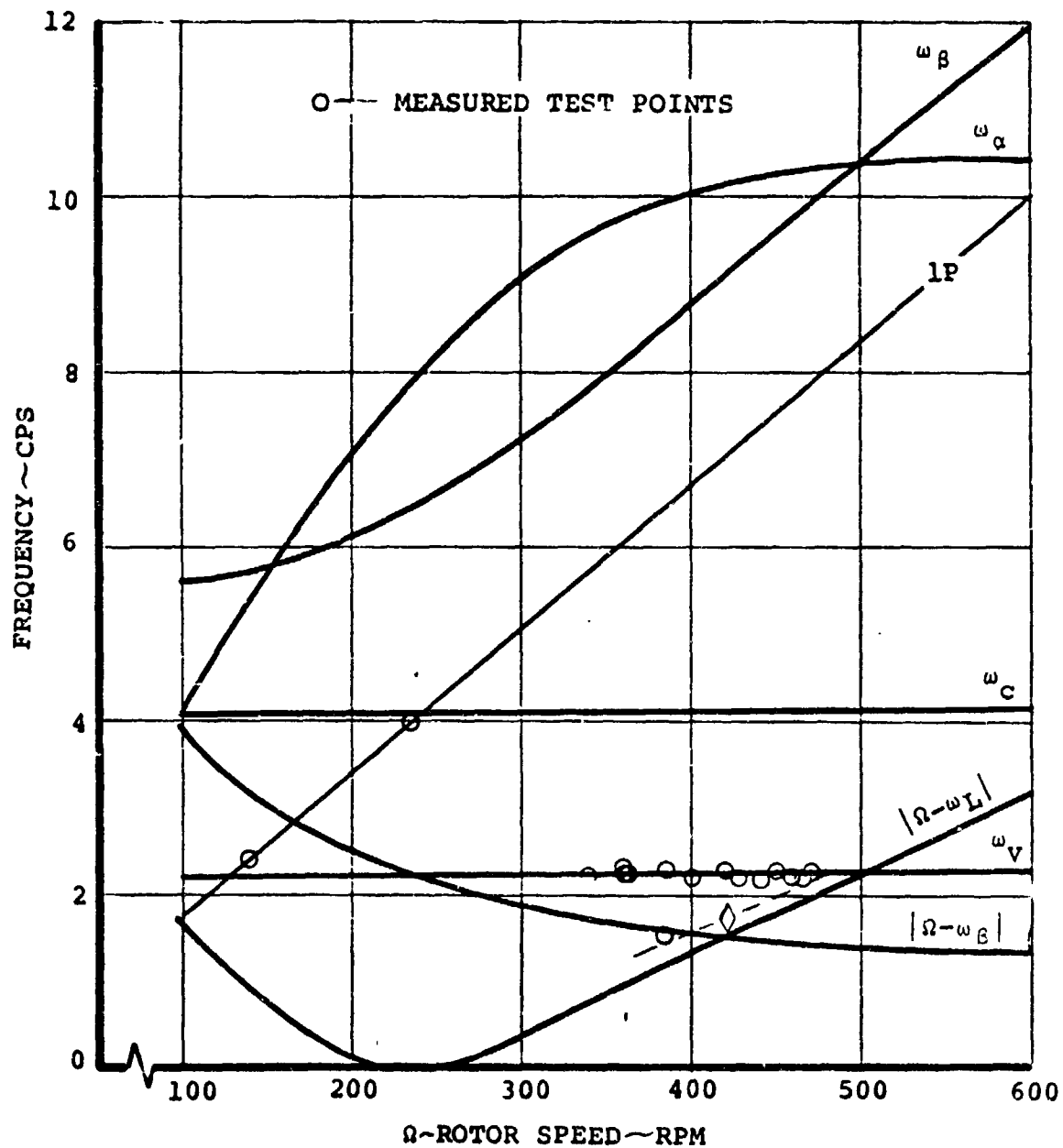
FIGURE 5.5.   26 FT. ROTOR - FULL STIFFNESS WING -
MODAL FREQUENCIES AT V = 100 KNOTS.

MODEL-222 FULL SCALE ROTOR TEST IN NASA
AMES 40 X 80 FOOT TUNNEL: FULL STIFF WING
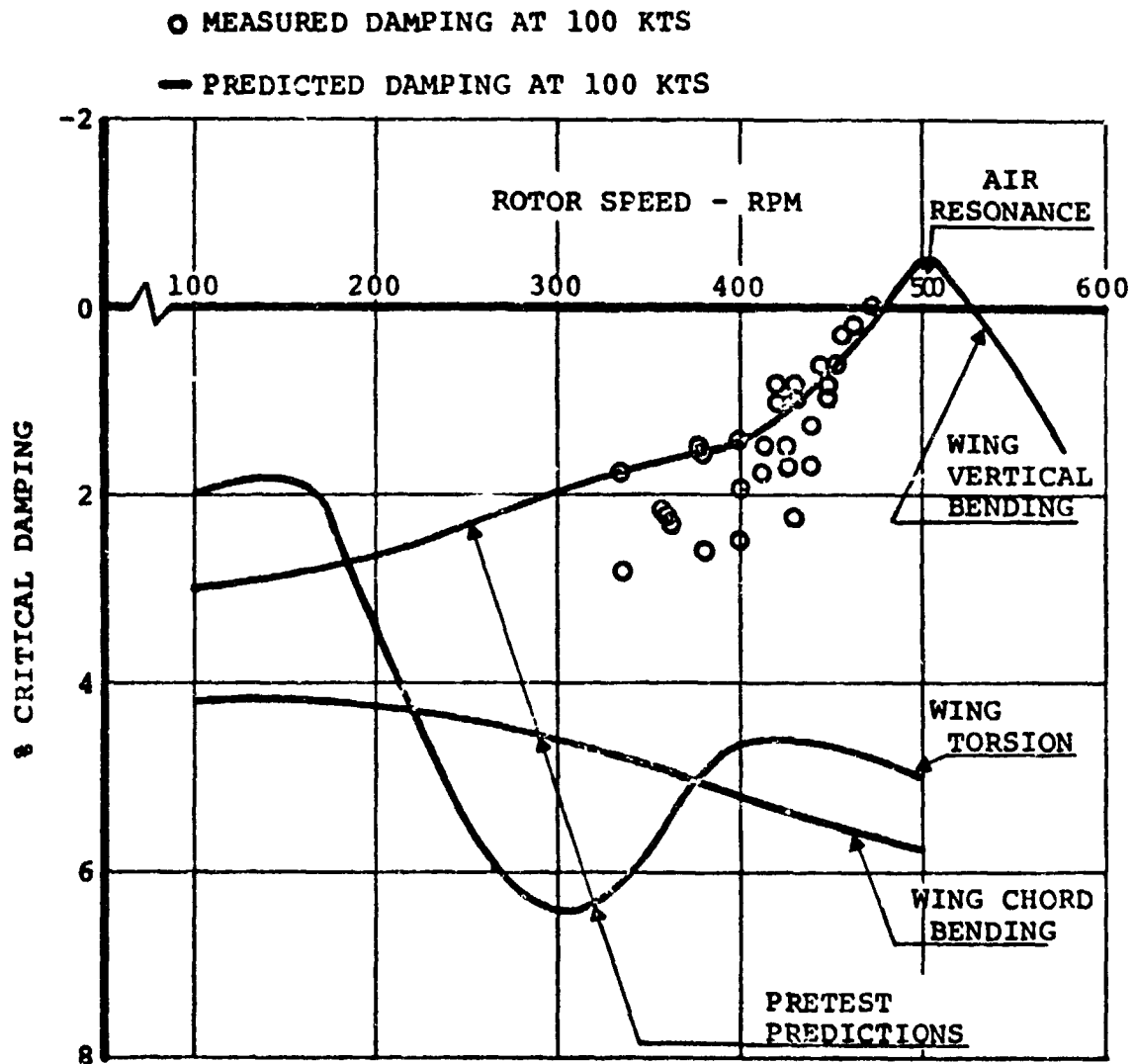
FIGURE 5.6.   CORRELATION OF PREDICTED AIR RESONANCE MODE
DAMPING AND MEASURED DAMPING OF THIS MODE
DURING TEST.   V = 100 KNOTS.

NASA AMES 40 X 80 WIND TUNNEL
TEST 410

150 KNOTS
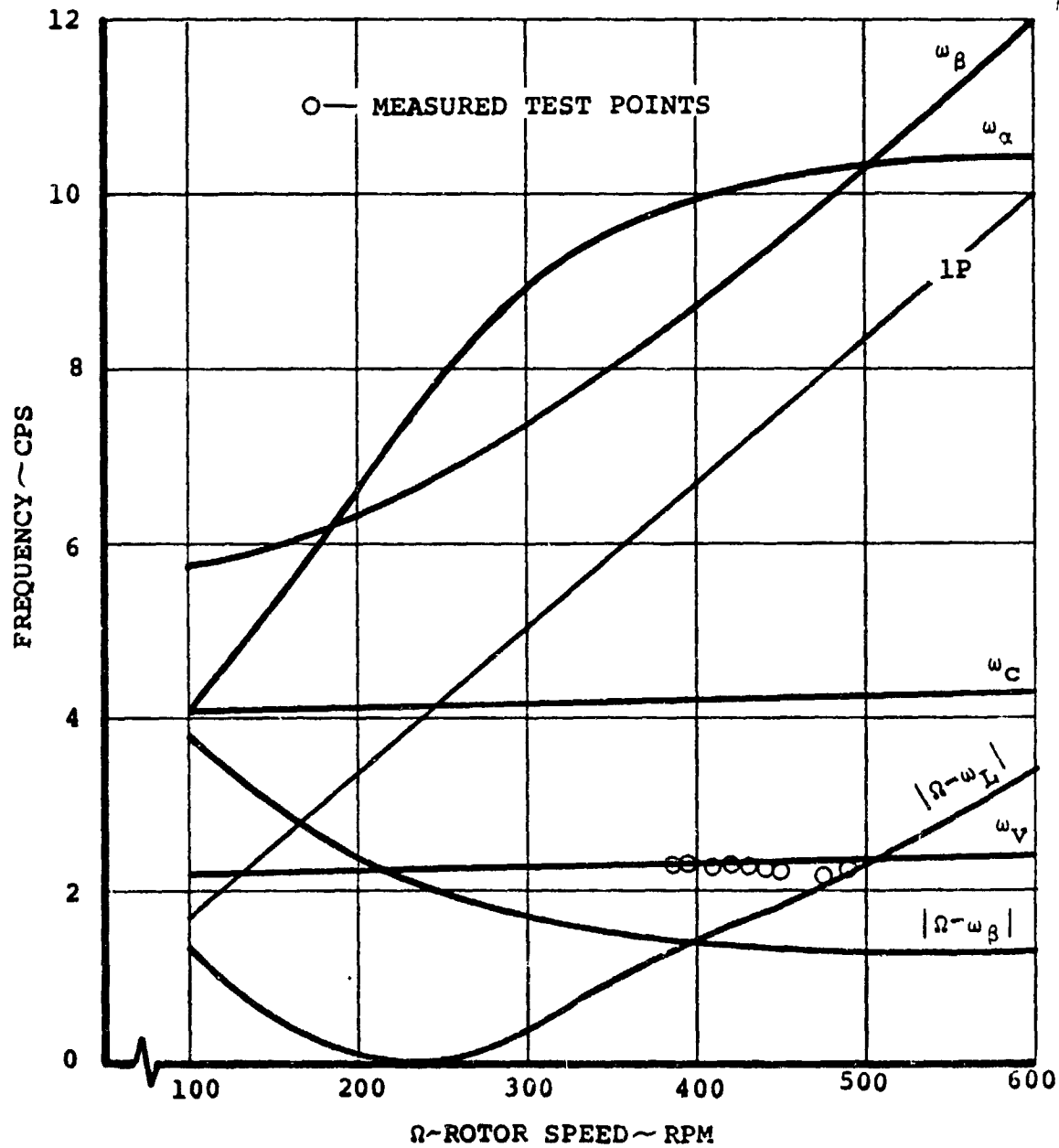
FIGURE 5.7. 26 FT. ROTOR - FULL STIFFNESS WING -
MODAL FREQUENCIES AT V = 150 KNOTS.

MODEL-222 FULL SCALE ROTOR TEST IN NASA
AMES 40x80 FOOT TUNNEL: FULL STIFF WING

O RUN 12 AT 140 KNOTS

━ PREDICTED DAMPING AT 150 KNOTS
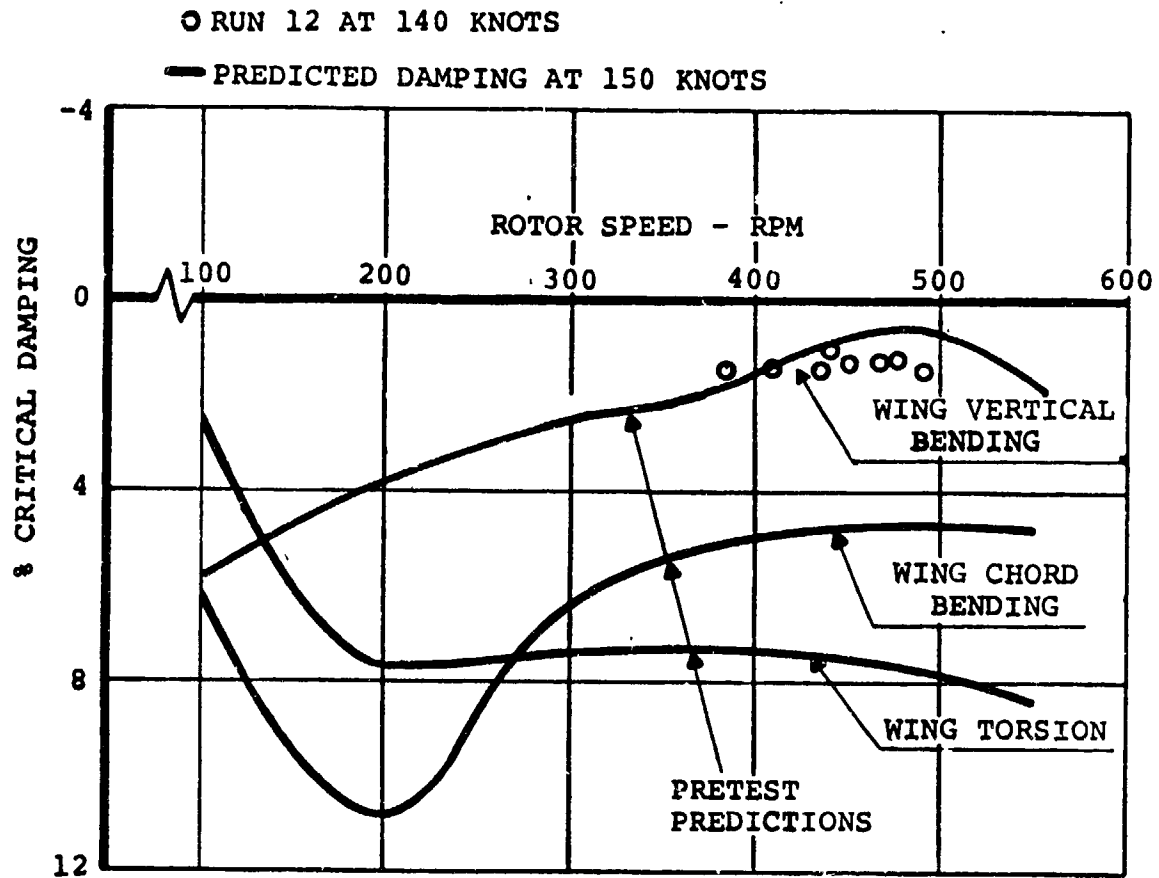
FIGURE 5.8. CORRELATION OF PREDICTED AIR RESONANCE MODE
DAMPING AND MEASURED DAMPING OF THIS MODE
DURING TEST. V = 140 KNOTS AND 150 KNOTS.

NASA AMES 40 X 80 WIND TUNNEL
TEST 410

200 KNOTS
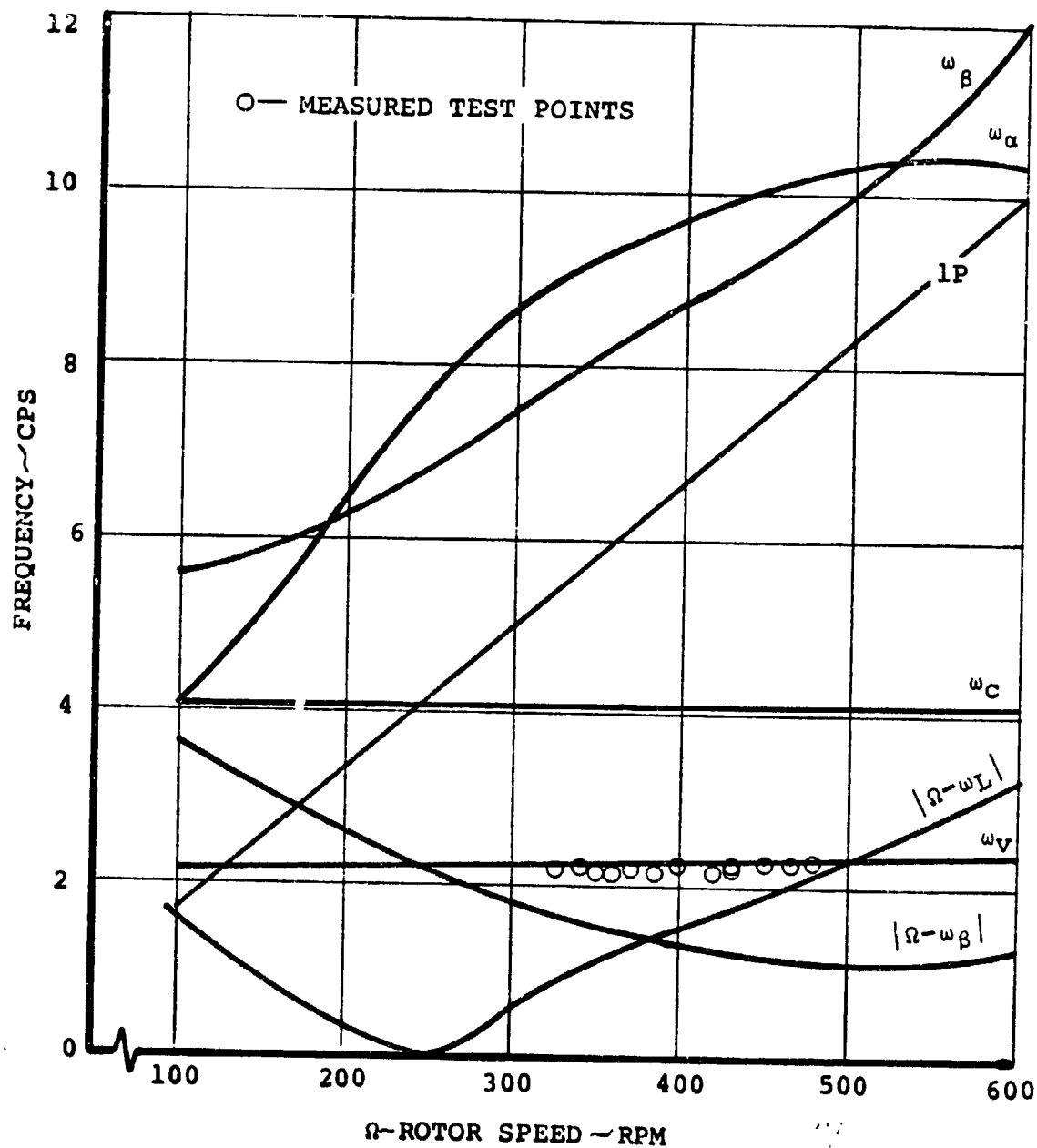
FIGURE 5.9. 26 FT. ROTOR - FULL STIFFNESS WING -
MODAL FREQUENCIES AT V = 200 KNOTS.

MODEL-222 FULL SCALE ROTOR TEST IN NASA
AMES 40x80 FOOT TUNNEL: FULL STIFF WING
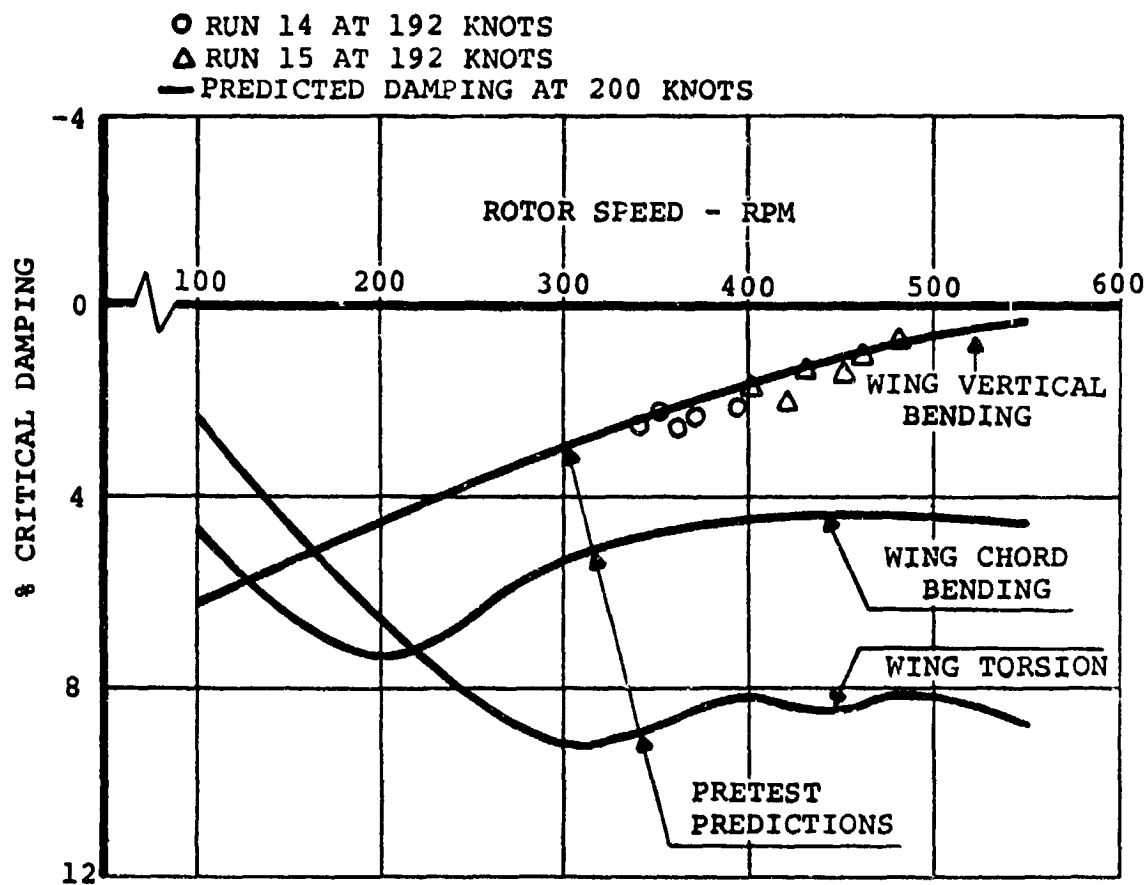
FIGURE 5.10. CORRELATION OF PREDICTED AIR RESONANCE MODE
DAMPING AND MEASURED DAMPING OF THIS MODE
DURING TEST.   V = 192 KNOTS AND 200 KNOTS.

is contained in Boeing Document D210-10435-1, Flying Qualities
and Aeroelastic Stability Analysis - C-48 User's Document.
A high degree of confidence also exists in the validity of the
feedback modelling as will be illustrated by correlation in
Paragraph 5.5.1.

5.4 INITIAL EXPLORATIVE STUDIES

During the initial work to augment the damping in the wing
vertical bending mode, four parameters were felt to be
important:

- O feedback signal
- O cyclic azimuth position
- O feedback loop gain
- O feedback loop time constant.

To better understand the general problem each one of these
parameters was studied in an elementary manner. The effect
on the wing vertical bending mode was noted along with the
effect on the wing torsion mode. The block diagram of the
feedback system is given in Figure 5.1.

Since the purpose of this system was to increase damping
in the wing vertical bending mode, the obvious choice for a
feedback signal was wing tip velocity. Experience with SAS
systems and general background knowledge of feedback control
systems indicated that damping may be increased in a mode by
feeding back the rate of change of the modal deflection -
wing tip vertical velocity. At the time it was felt to be
unnecessary to investigate other feedback signals. (More

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about this in Section 5.5).

Cyclic azimuth position was the second parameter to be considered. It was felt that there was an optimum orientation of the hub forces and moments which would yield the most damping in the wing vertical bending mode without destabilizing other modes. Orientation of the hub forces and moments was obtained by inputting various amounts of A_1 and B_1 cyclic pitch (Figure 5.11) per unit feedback signal. Thus, by varying the ratio of A_1 to B_1 the force and moment orientation was moved around the azimuth. It was subsequently realized that this procedure also controlled the net phase of the feedback signal. To investigate azimuth variation effect, carpet plots with A_1 and B_1 gain variations were made.

Feedback loop gain was expected to control the level of damping attained. The gain was therefore increased until either the damping no longer increased or some other mode was driven unstable. As discussed below, the torsion mode was driven unstable and thus defined the maximum level of gain.

Feedback loop time constants (τ) of 0., .05 and .1 seconds were investigated to yield some insight into the effect of actual feedback system hardware (actuators, sensors, etc.) on the dynamics of the overall system. These values of τ were chosen because experience indicated that control system time constants varied between .05 and .1 seconds.

FIGURE 5.11. SIGN CONVENTION FOR AEROELASTIC
STABILITY INVESTIGATIONS

Several different areas of the flight envelope were explored to evaluate the effect of different velocity and RPM combinations. Major emphasis was placed on the cruise flight conditions (386 RPM; 100, 150 and 200 knots) and some study has been made of corner conditions of the flight envelope. Discussion of the individual conditions follows.

5.4.1 Discussion of Results of Initial Study

Perfect feedback ($\tau = 0.0$ seconds) was evaluated for the flight conditions of 386 RPM, 100 knots and 386 RPM, 200 knots. Variations in A_1 and B_1 gain were made and the only modes significantly effected were the wing vertical bending and wing torsion modes (Figures 5.12, 5.13, 5.14 and 5.15). Gains are expressed in degrees of cyclic per foot per second. Significant increases in wing vertical bending damping are seen in both conditions. At maximum gain at 100 knots 32% critical damping is achieved; at 200 knots 80% critical damping is achieved. However, at these gains the wing torsion mode is driven unstable in both cases. Figures 5.16 and 5.17 define the gain envelope for stability of both the wing vertical bending and torsion modes. Using this criteria the designer may pick values of gain which give him a satisfactory solution. For instance at 200 knot values of a = .4 and b = -.4 yield a decrease in torsion damping from 1.8% to .9% but a dramatic increase in vertical bending damping from about 1% to 20%. If a further increase in damping is required in the vertical bending mode and it is desired to have no

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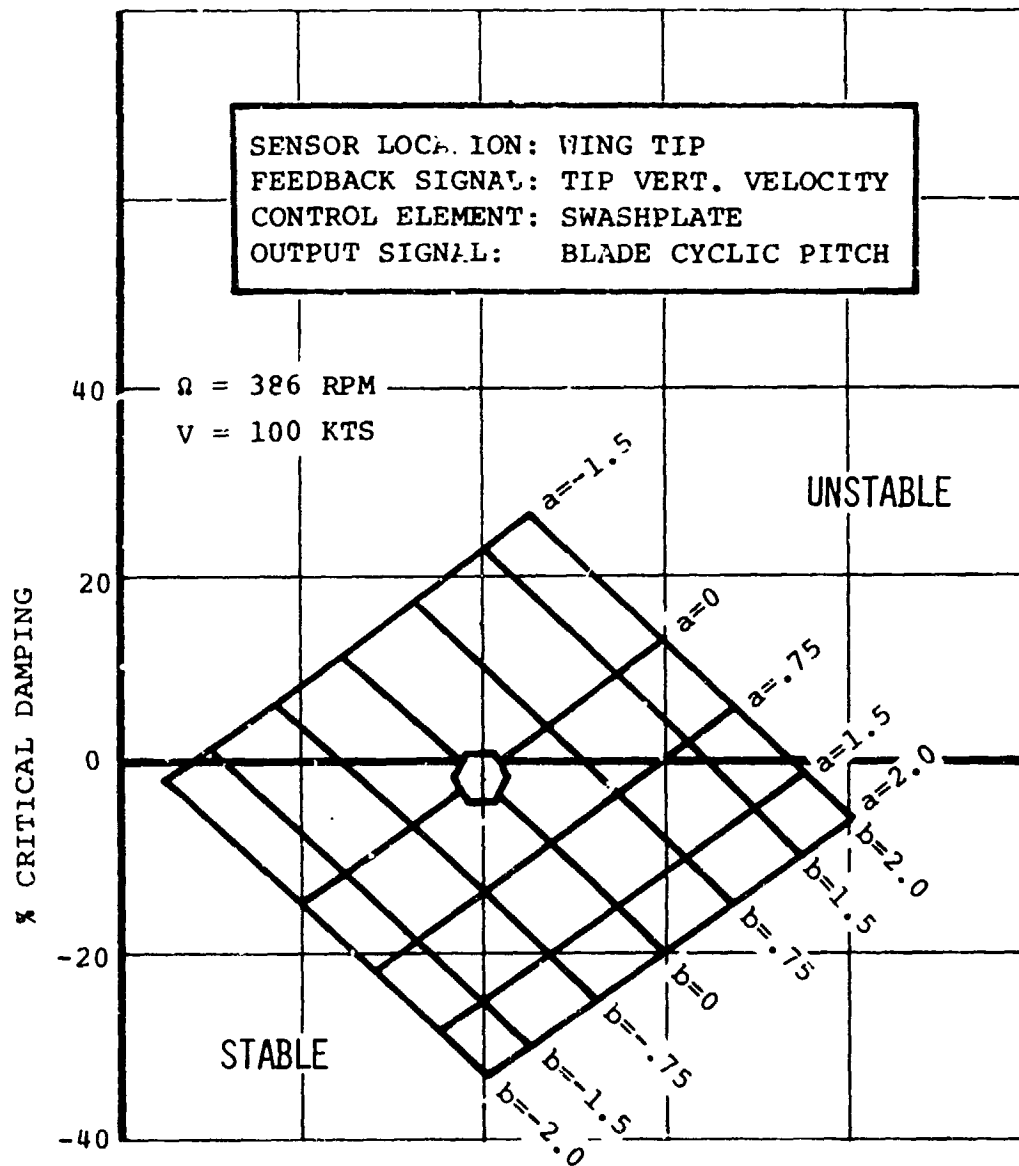
FIGURE 5.12 FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE
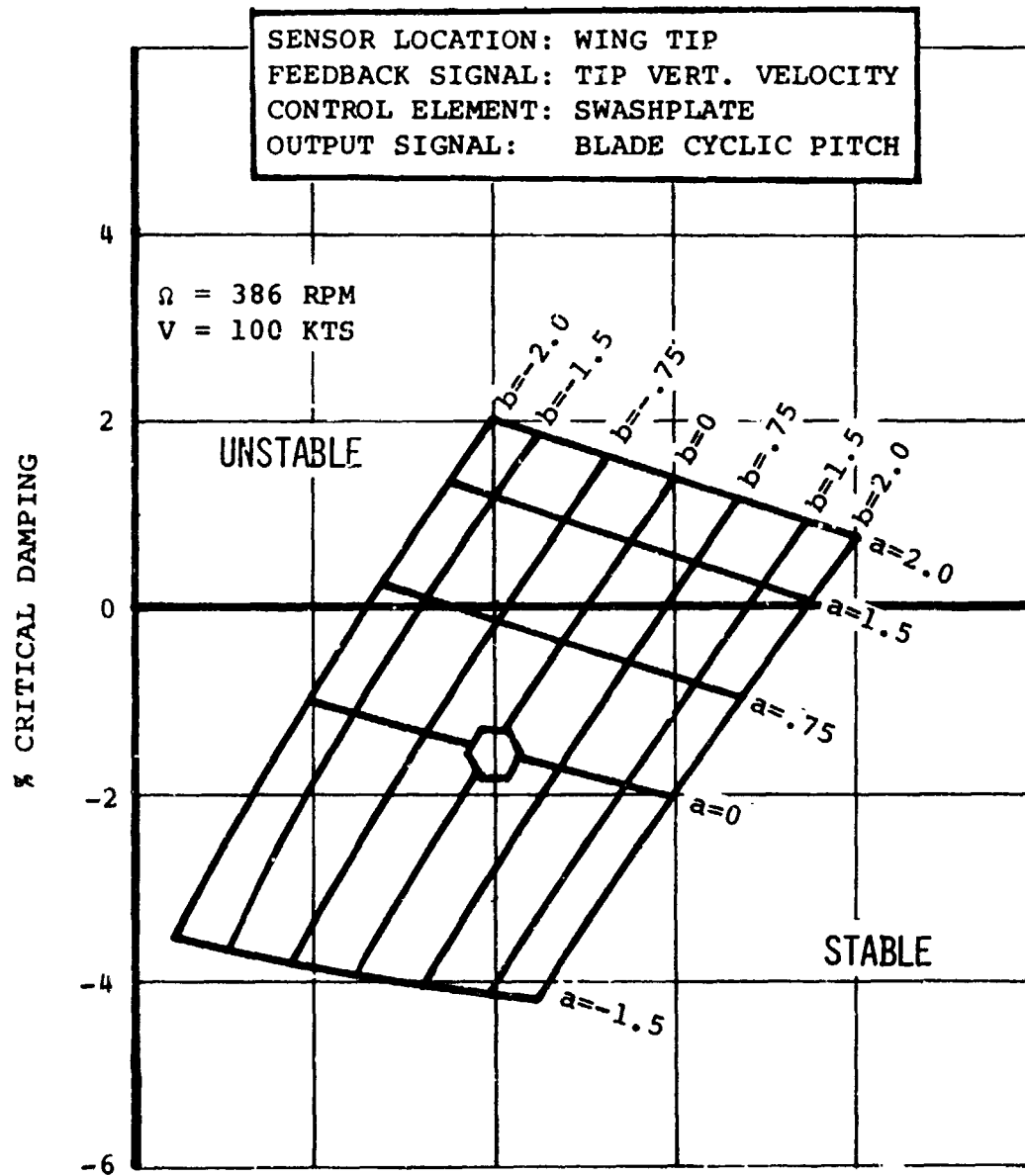
FIGURE 5.13 FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE
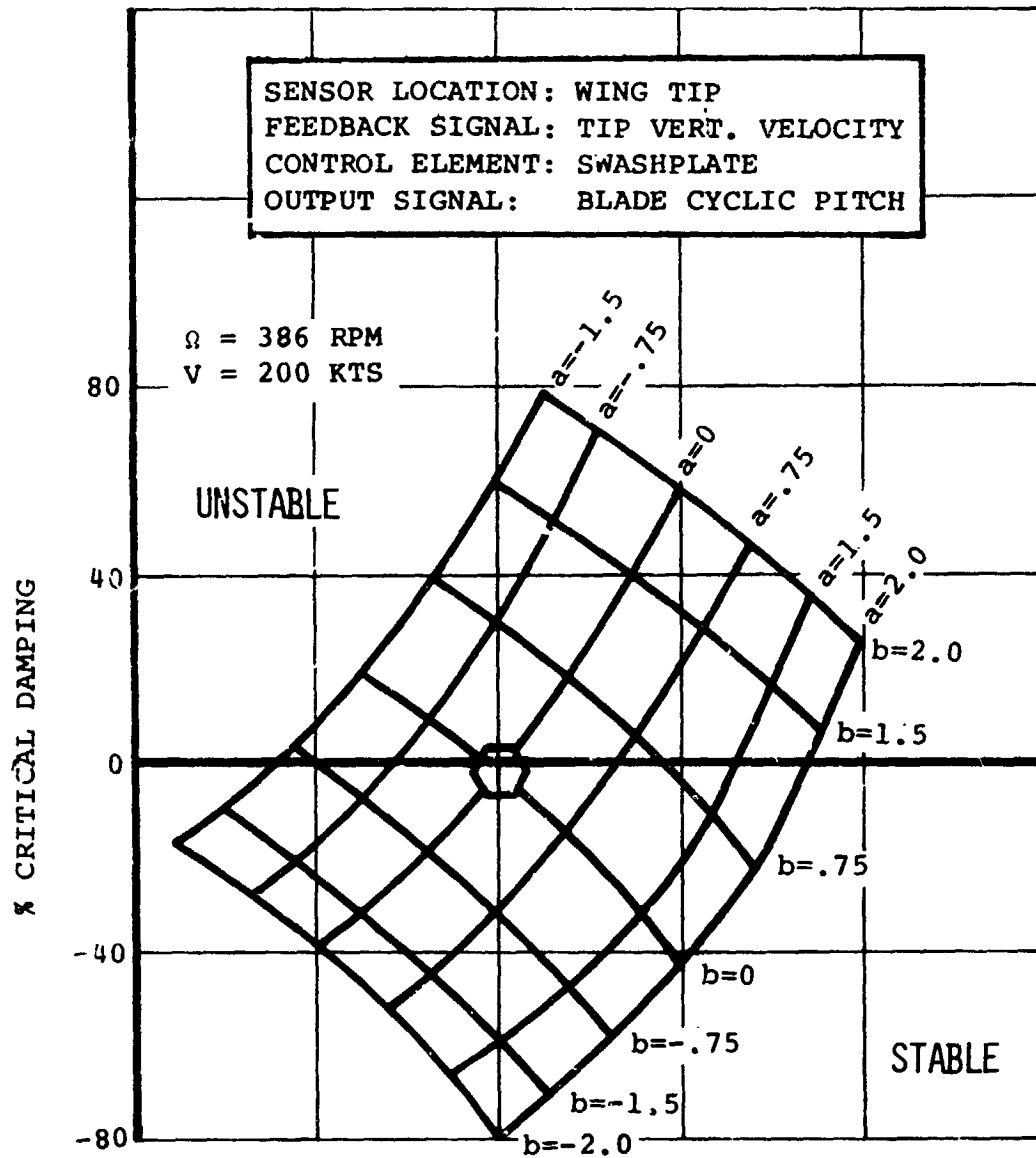
FIGURE 5.14 FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE
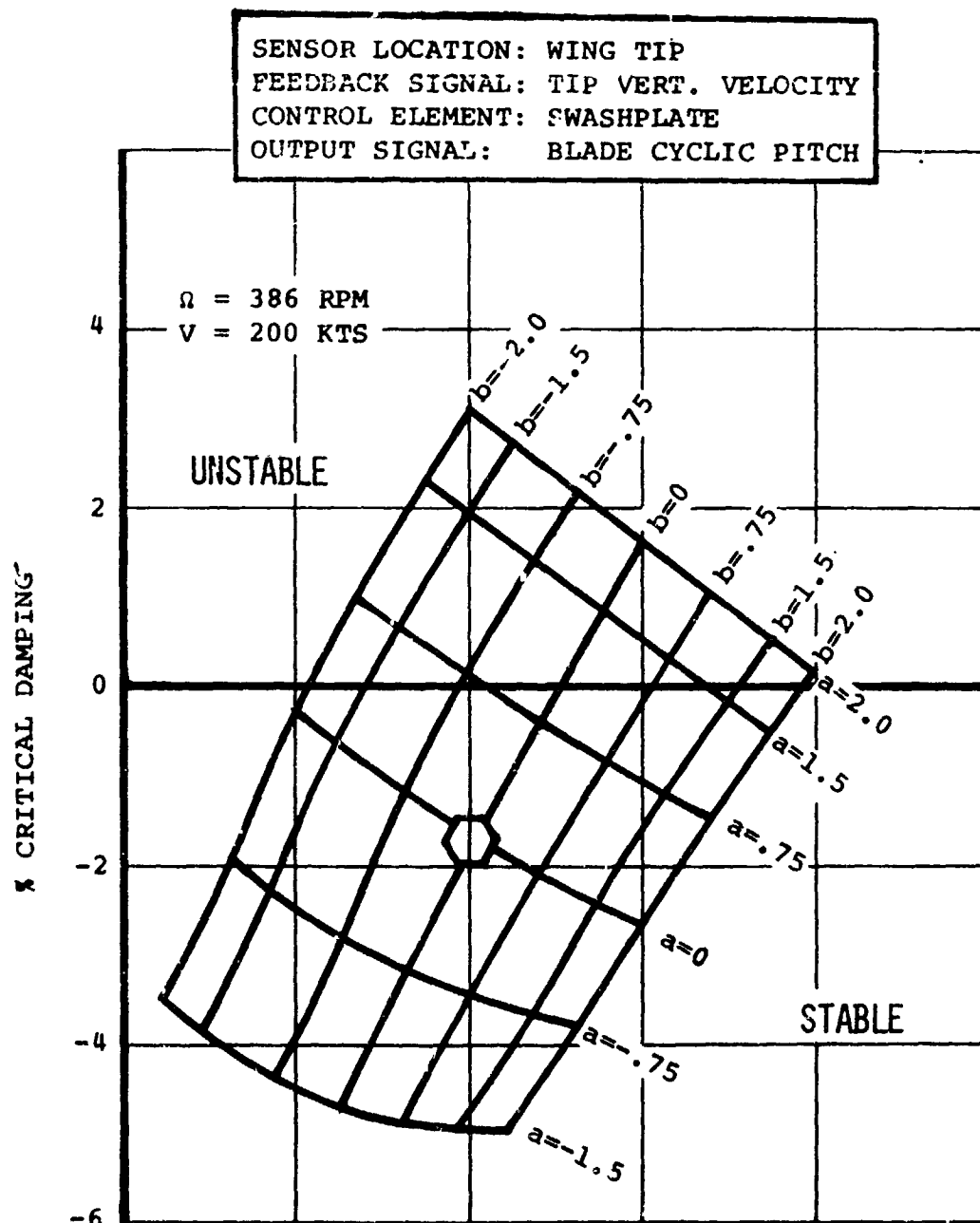
FIGURE 5.15 FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE

SENSOR LOCATION: WING TIP
FEEDBACK SIGNAL: TIP VERT. VELOCITY
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH
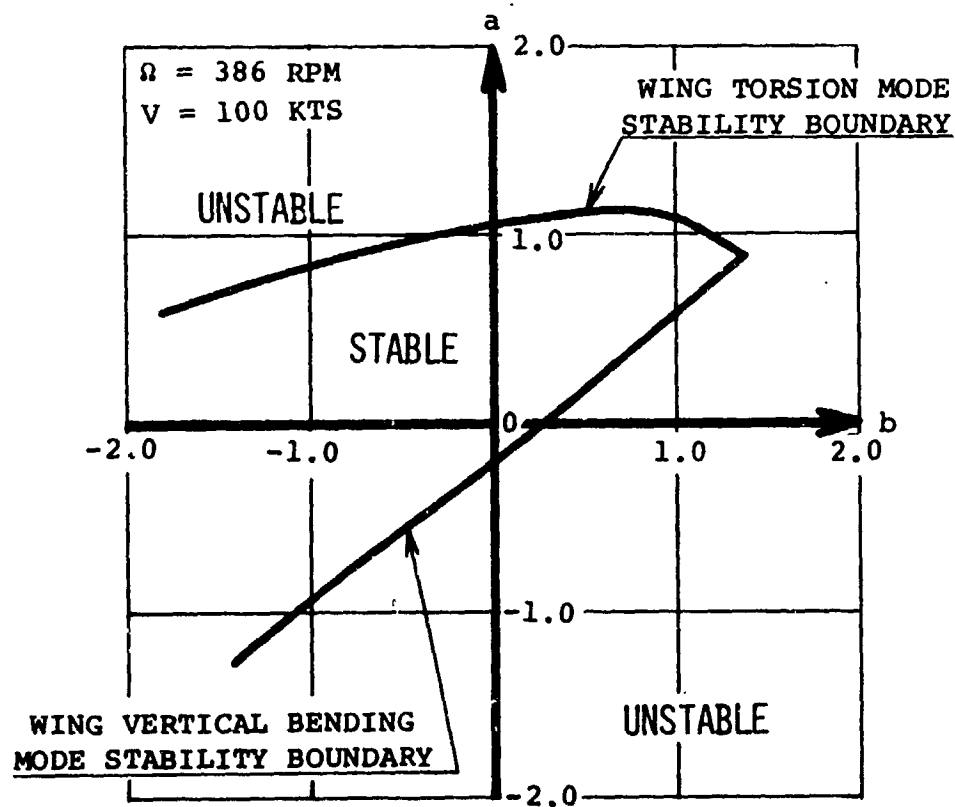
FIGURE 5.16 FEEDBACK EFFECT ON MODAL DAMPING OF
              STABILITY FOR FEEDBACK COMBINATIONS

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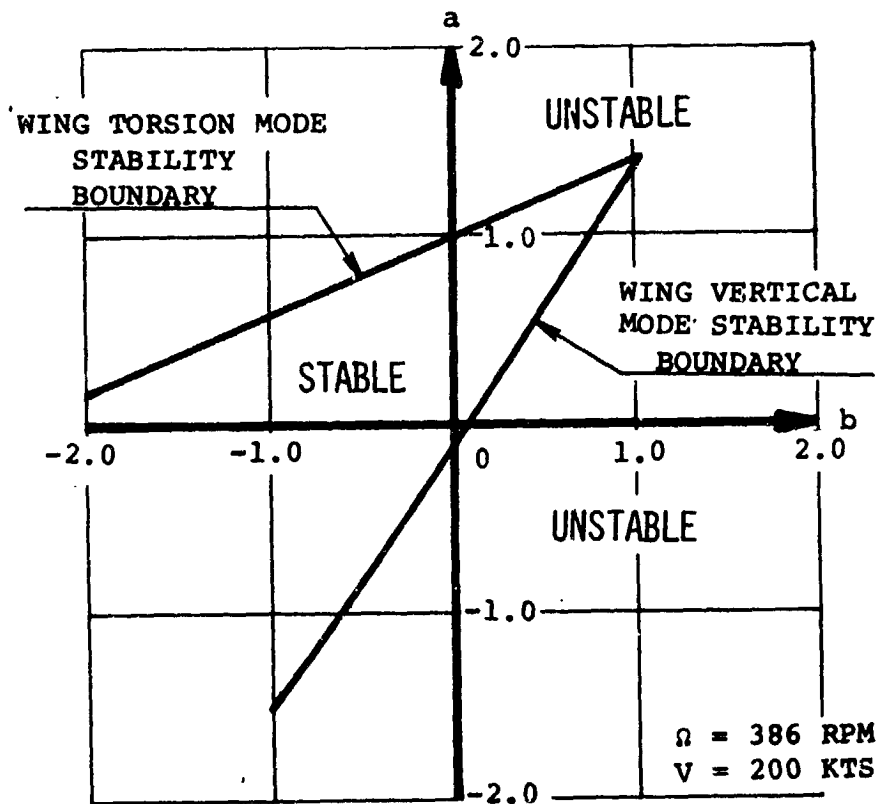
FIGURE 5.17 FEEDBACK EFFECT ON MODAL DAMPING OF
STABILITY FOR FEEDBACK COMBINATIONS

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effect on the wing torsion mode a more complex analysis must be performed as discussed in Section 5.5.

5.4.2 Ideal System With First Order Time Lag

Control system time constants were evaluated at the flight condition of 386 RPM and 150 knots. Figures 5.18, 5.19, 5.20 and 5.21 show the effect of gain at time constants of .05 and 0.1 seconds on the wing vertical bending and torsion modes. Stability root migration with gain is much the same as with zero time constant; however, as seen in Figures 5.22 and 5.23 the effect of time constant is to reduce effectiveness. The vertical bending mode is more affected by τ than the torsion mode. Damping is decreased 11% in the vertical bending mode while there is only a .5% change in the wing torsion mode. Since there is such a large variation in the modal response of interest it behooves the analyst to carefully evaluate the dynamic characteristics of each piece of system hardware and include this information in his math model.

5.4.3 Variations With RPM

Evaluation of the feedback system at 500 RPM and 100 knots was made (Figures 5.24, 5.25, and 5.26). Feedback with zero time constants was investigated for a range of A_1 and B_1 gain.

The results are similar to those at 386 RPM and indicate no drastic changes in system operations with RPM.

An extreme 200 RPM condition was investigated at velocities of 100 and 200 knots (Figures 5.27 through 5.32). The same trends are evident here as at 386 RPM and 500 RPM. The gain
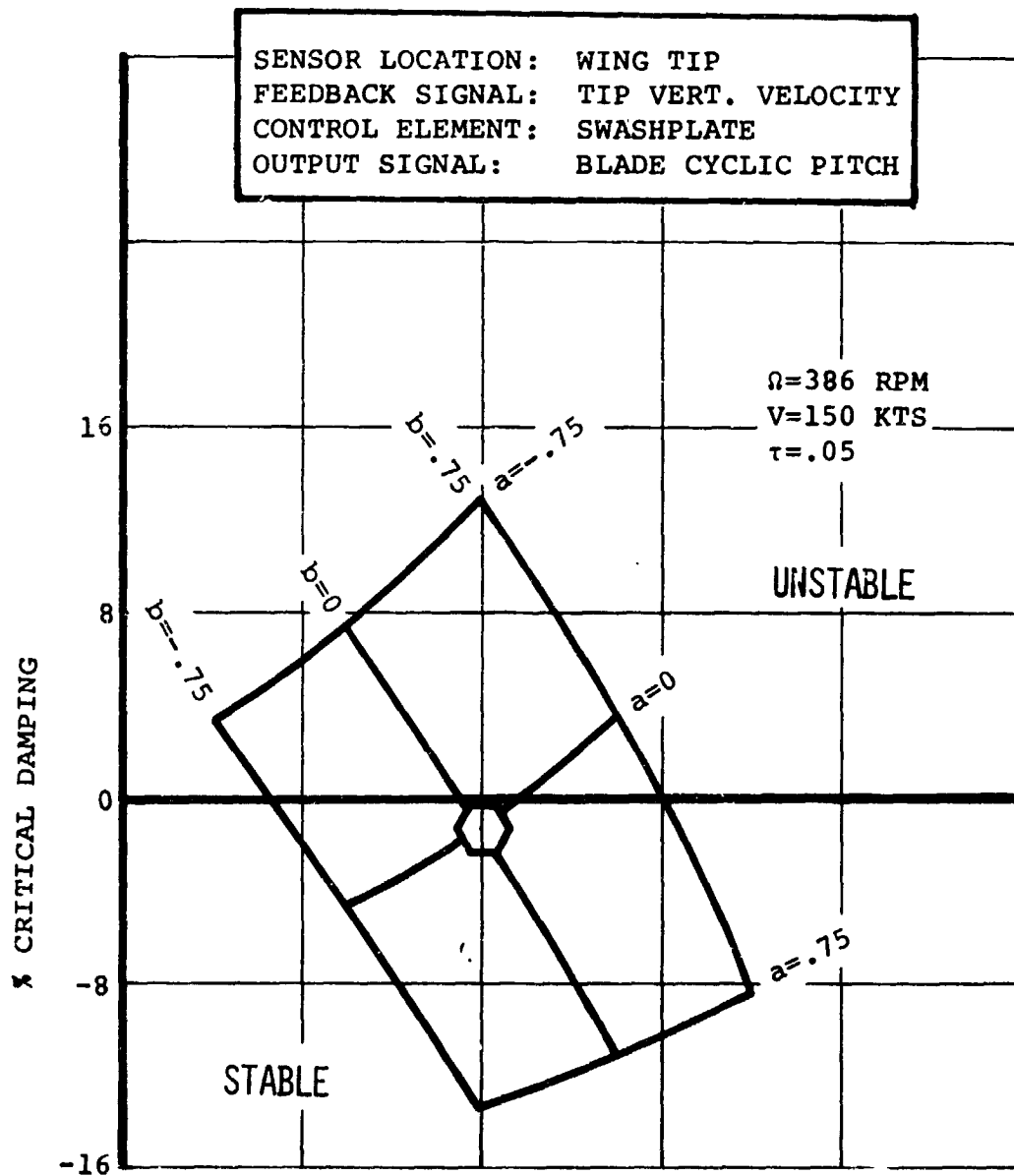
FIGURE 5.18. FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE WITH TIME CONSTANT
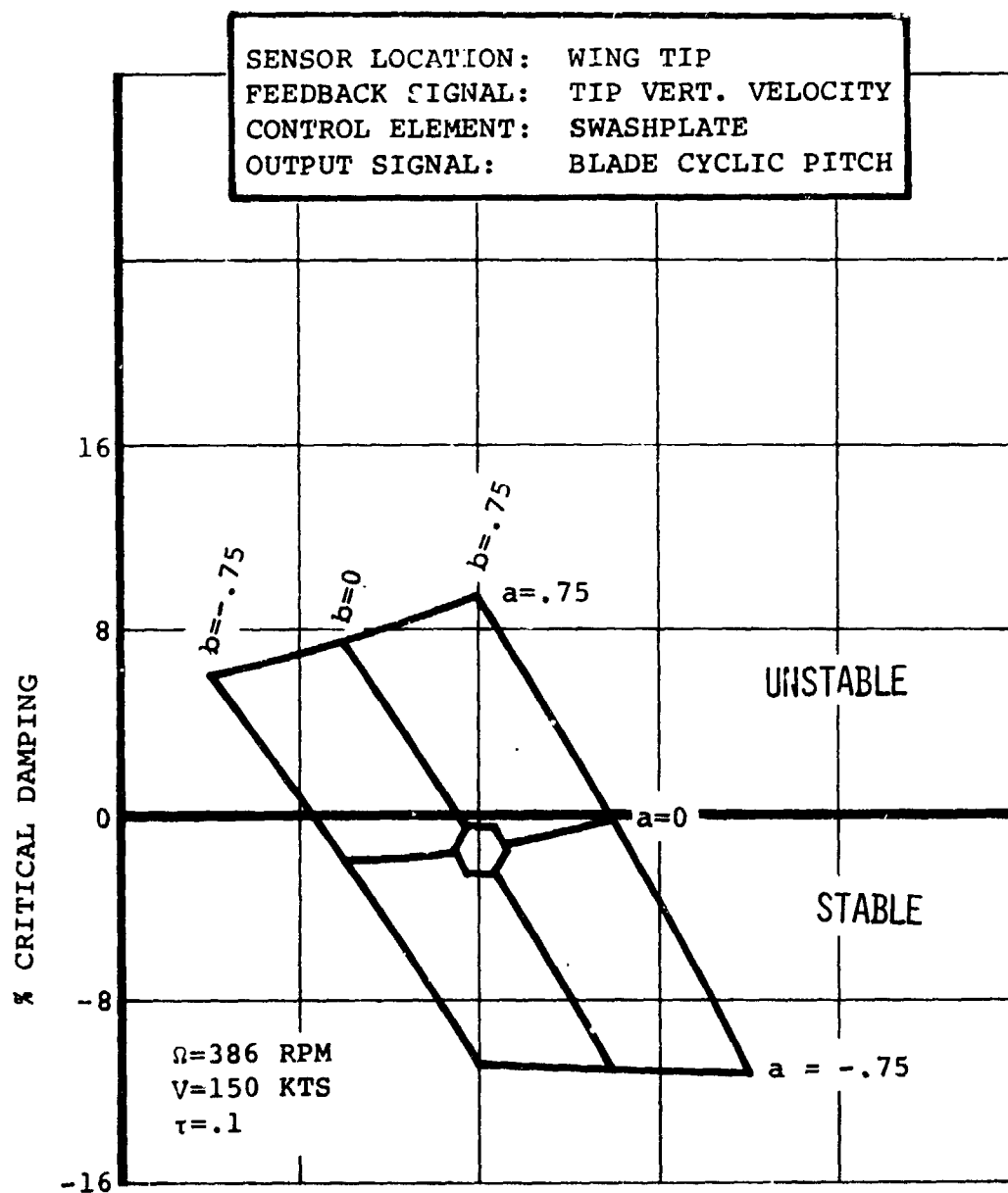
FIGURE 5.19. FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE WITH TIME CONSTANT
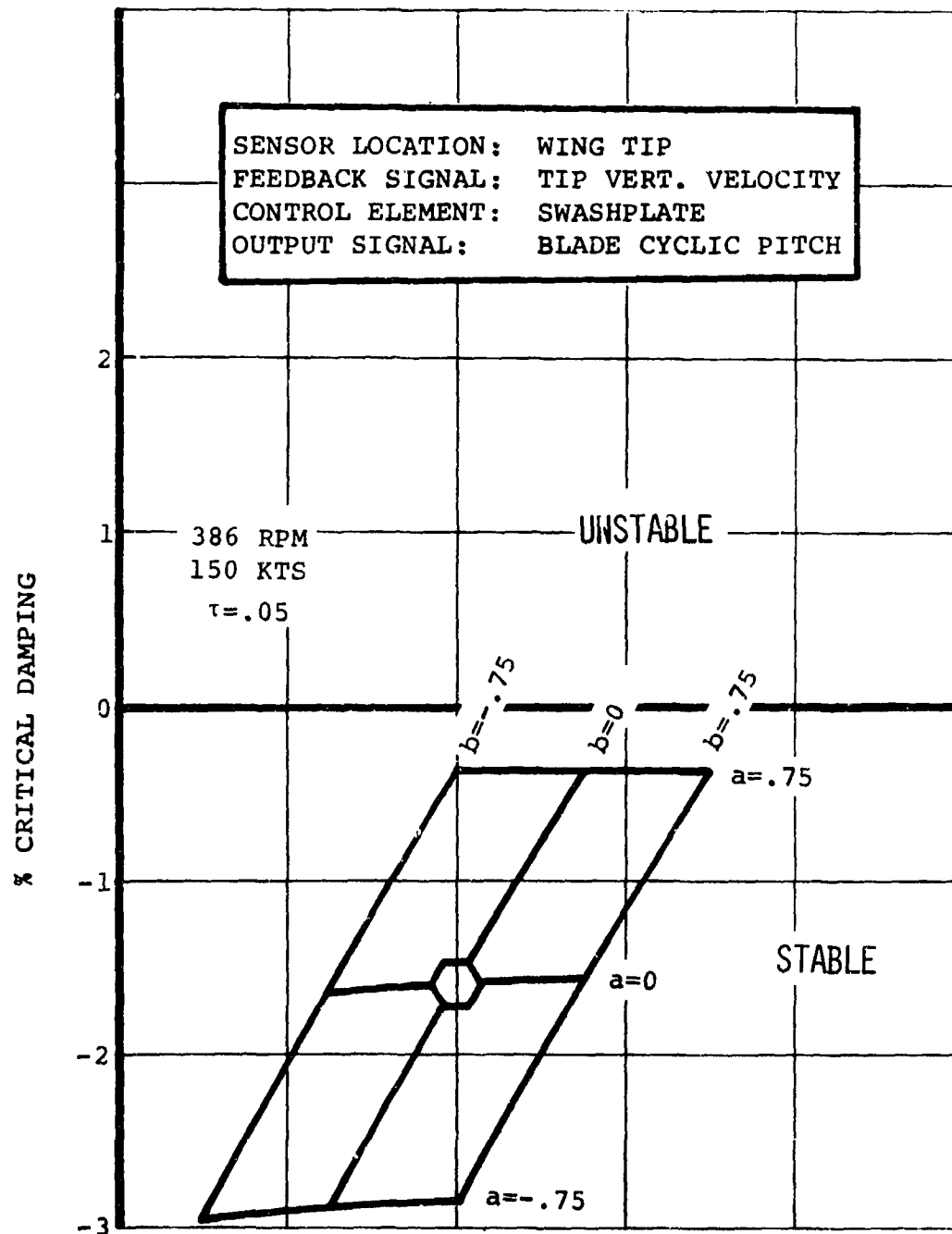
FIGURE 5.20. FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE WITH TIME CONSTANT
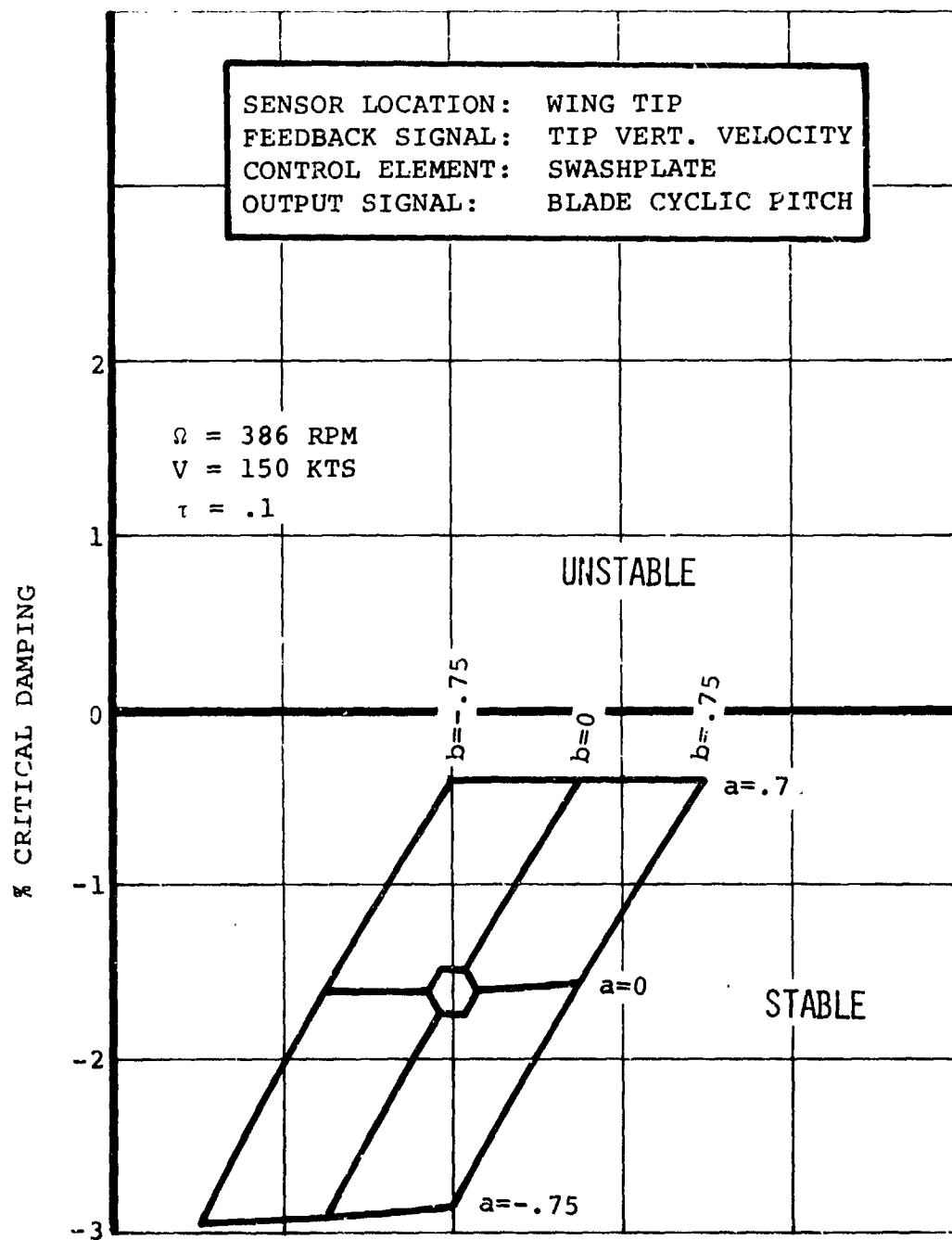
FIGURE 5.21. FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE WITH TIME CONSTANT
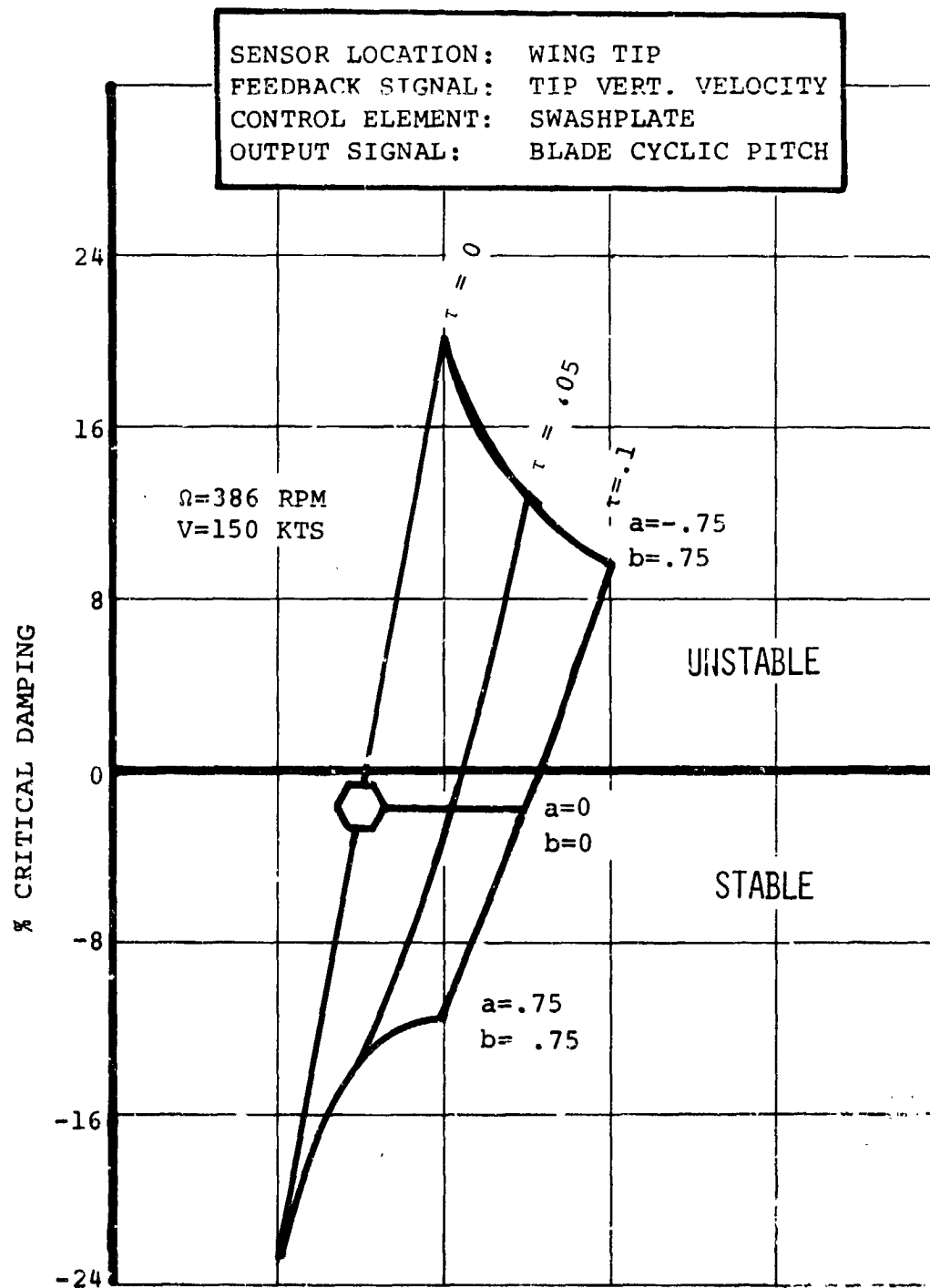
FIGURE 5.22. FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE WITH TIME CONSTANT

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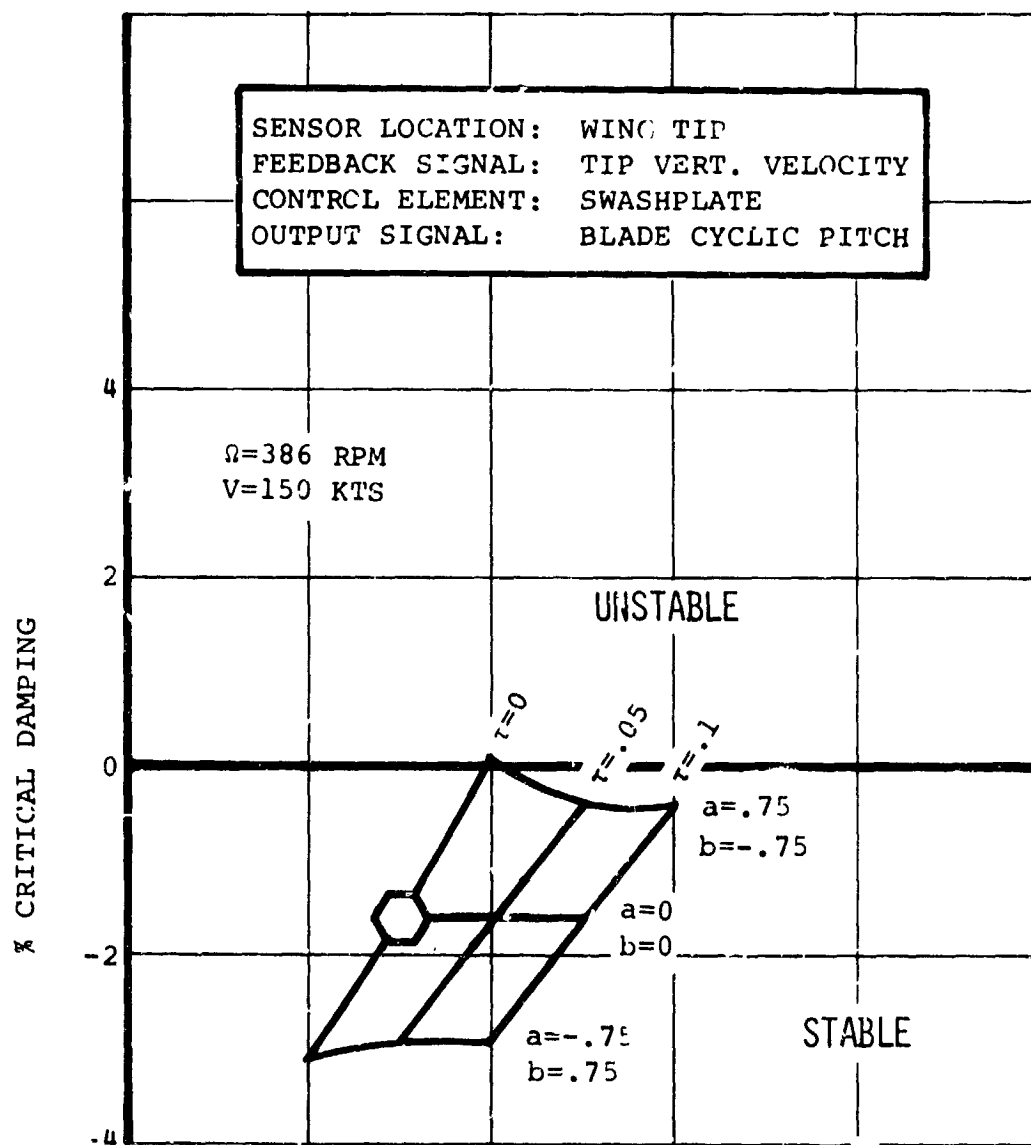
FIGURE 5.23. FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE WITH TIME CONSTANT
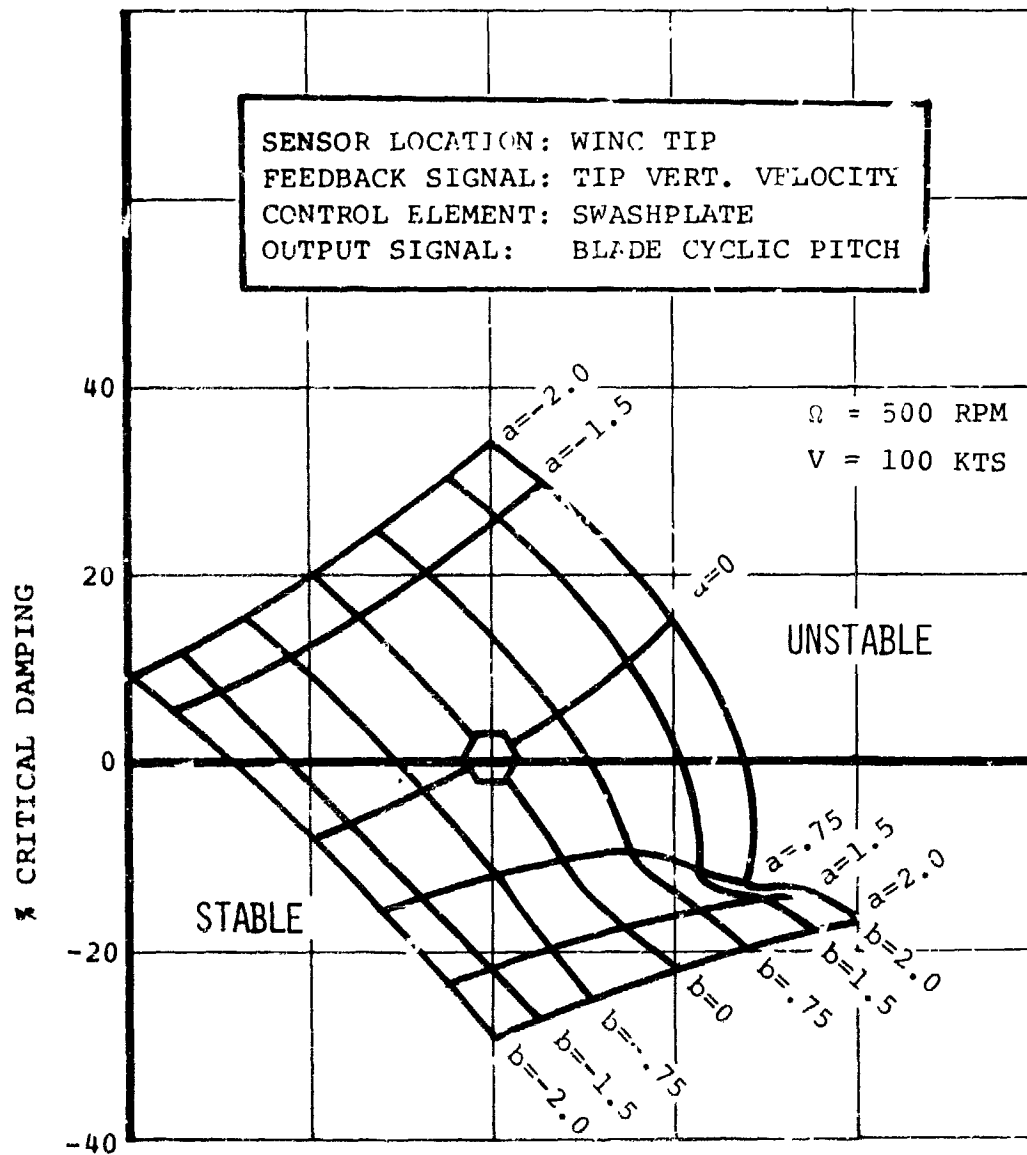
FIGURE 5.24. FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING MODE
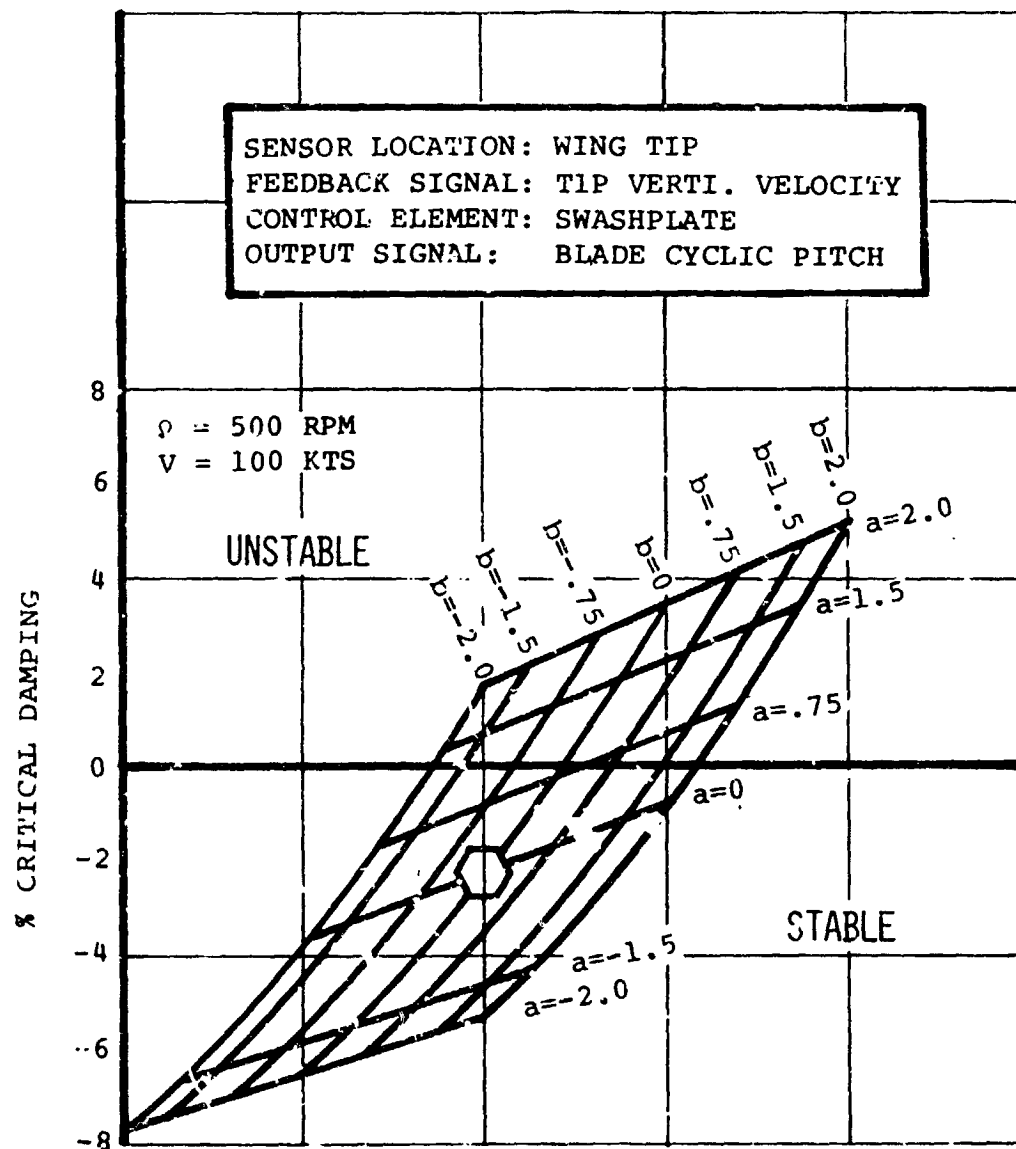
FIGURE 5.25 FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE

SENSOR LOCATION: WING TIP
FEEDBACK SIGNAL: TIP VERT. VELOCITY
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH
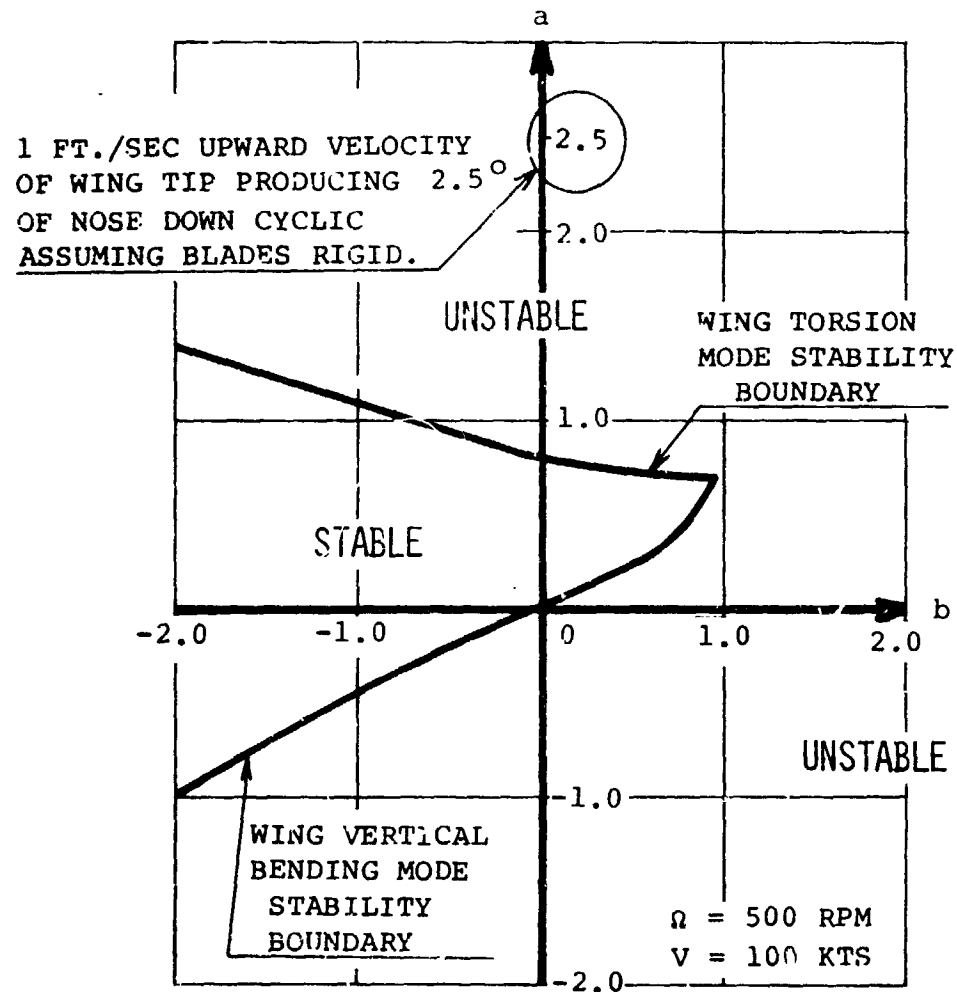
FIGURE 5.26 FEEDBACK EFFECT ON MODAL DAMPING OF
STABILITY FOR FEEDBACK COMBINATIONS

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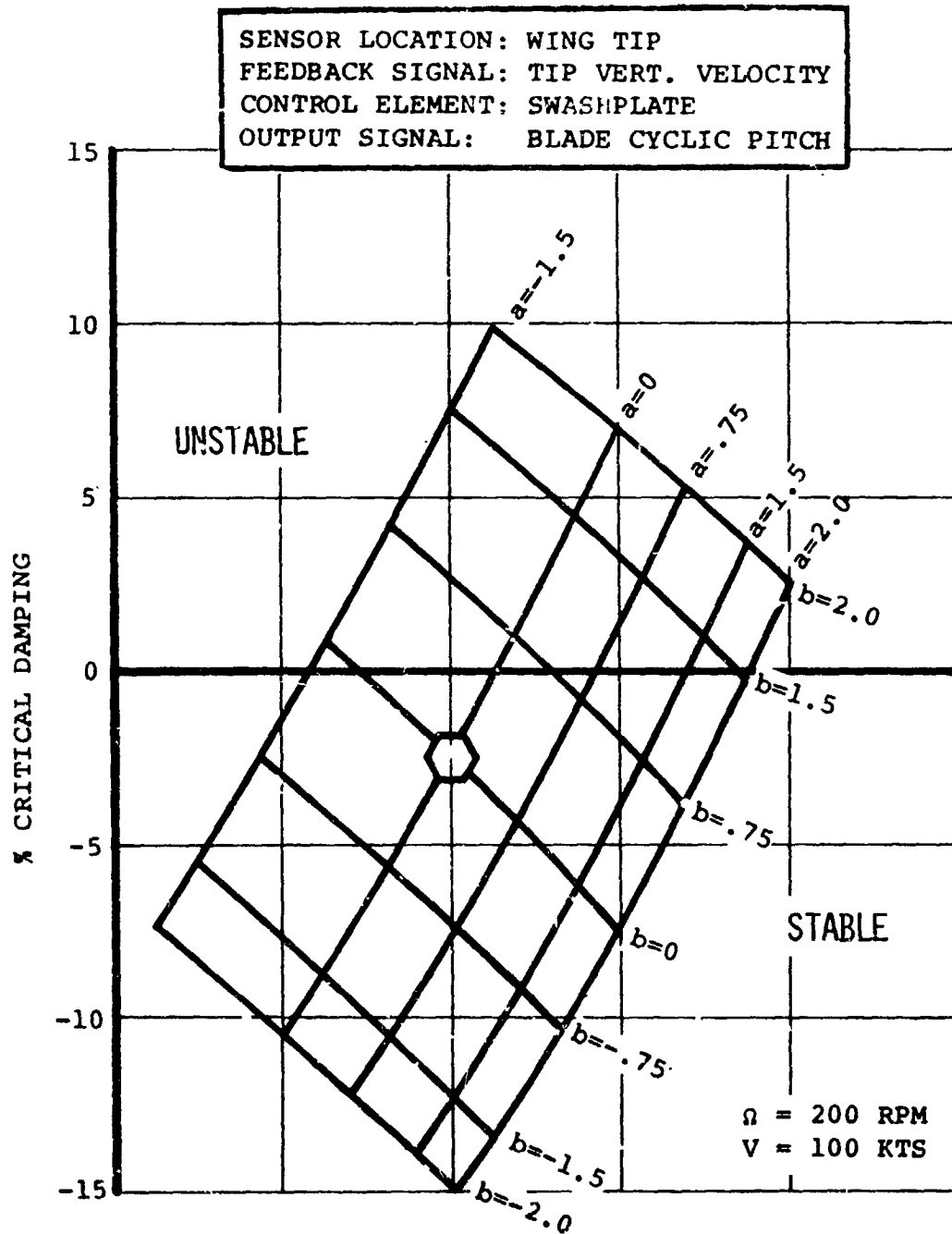
FIGURE 5.27 FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING
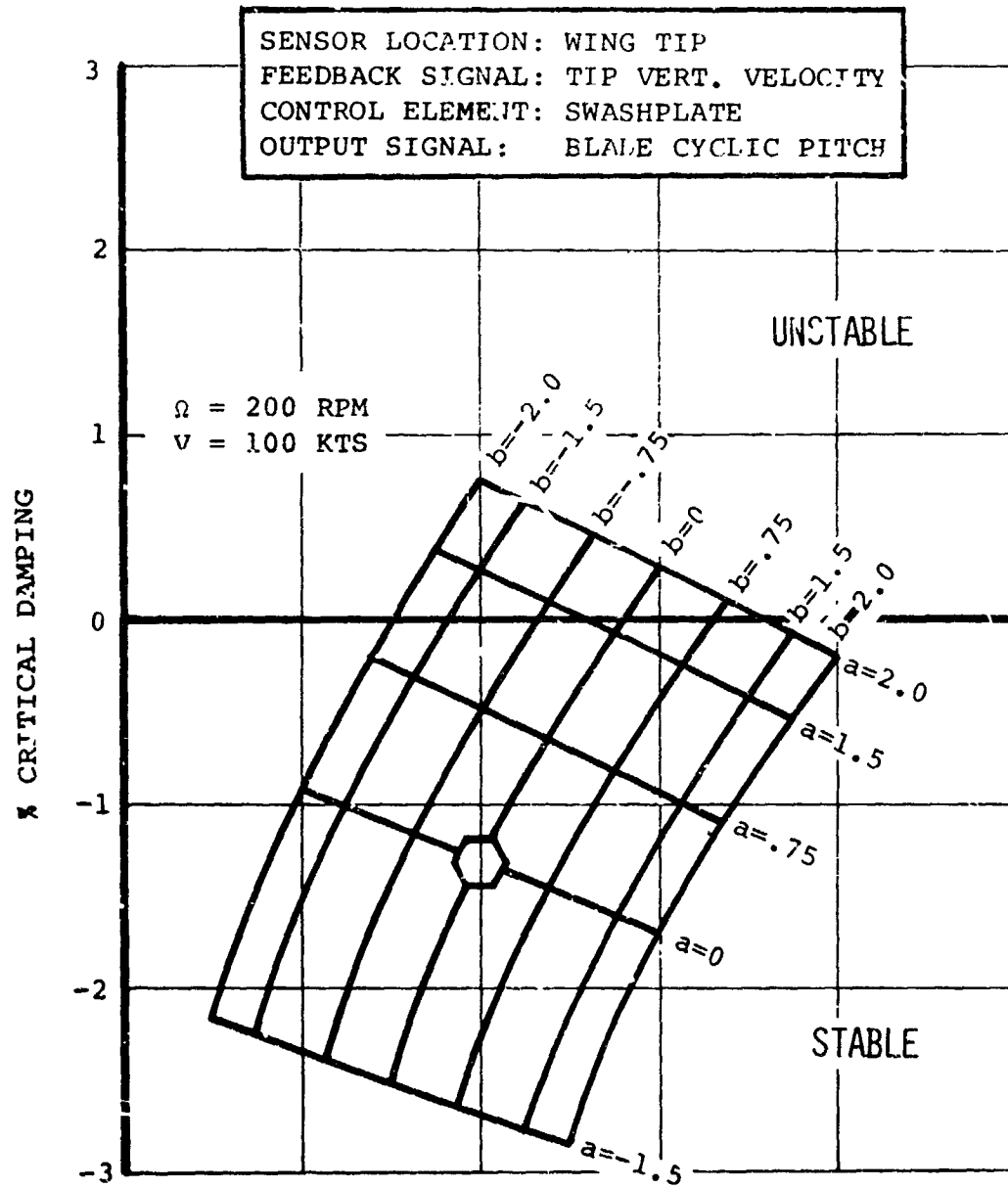
FIGURE 5.28 FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE
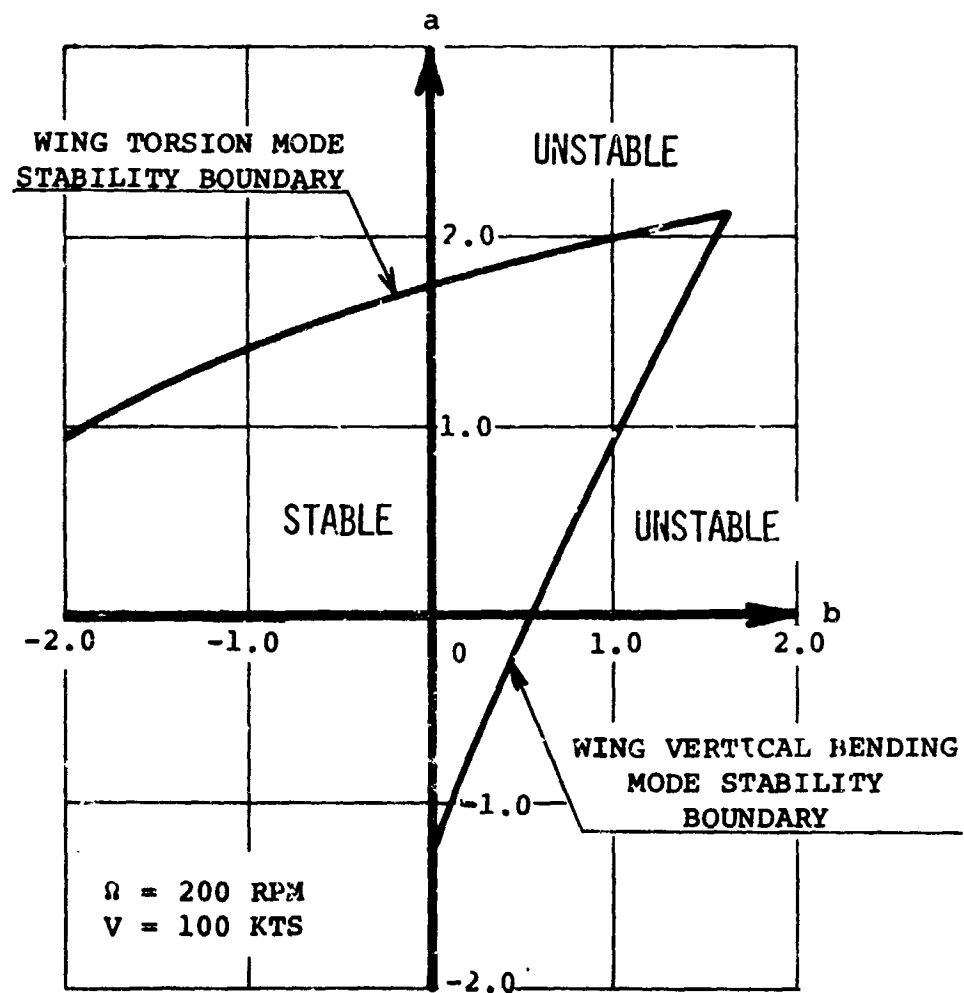
FIGURE 5.29 FEEDBACK EFFECT ON MODAL DAMPING OF AIRCRAFT
STABILITY FOR FEEDBACK COMBINATIONS
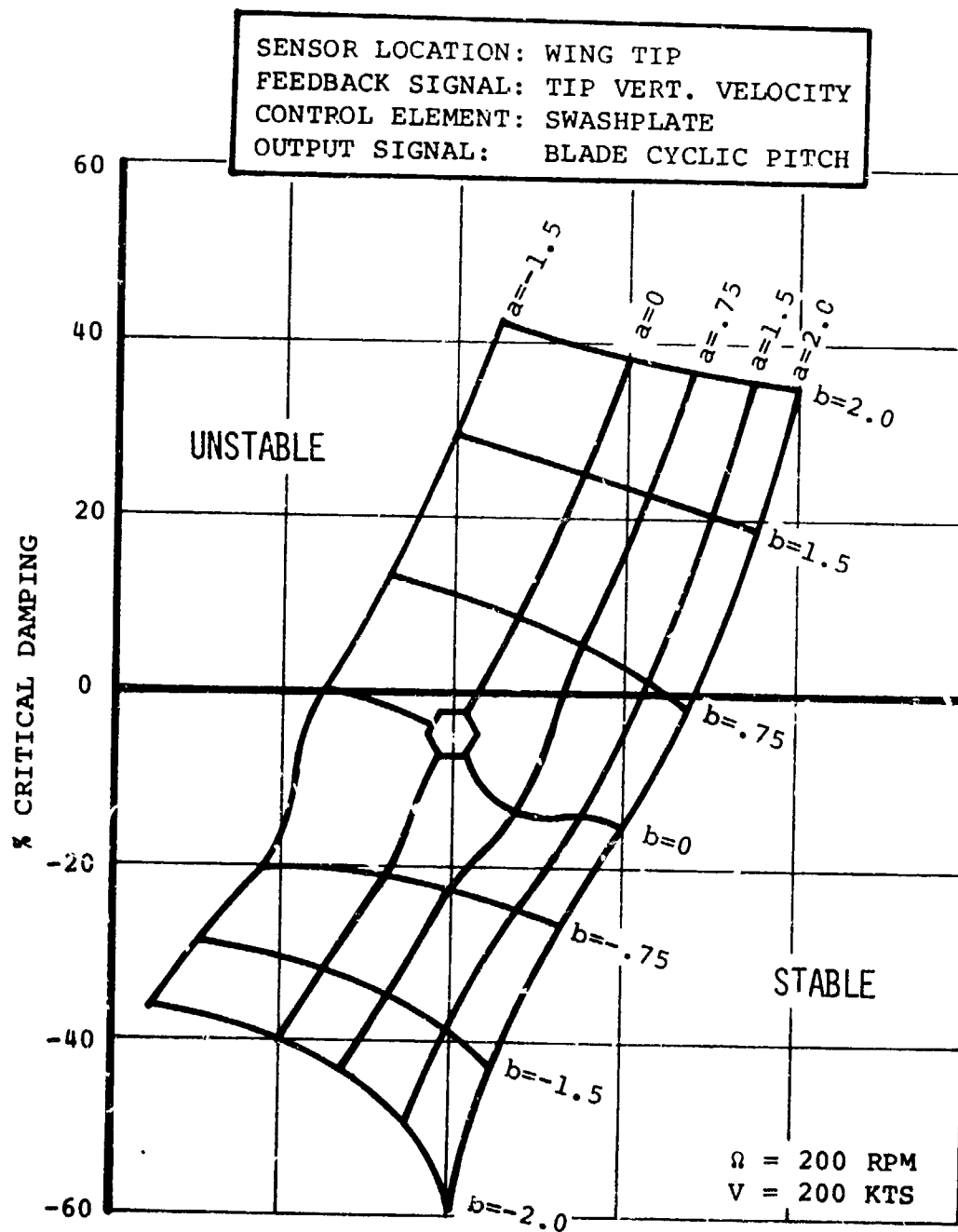
FIGURE 5.30 FEEDBACK EFFECT ON MODAL DAMPING OF
WING VERTICAL BENDING
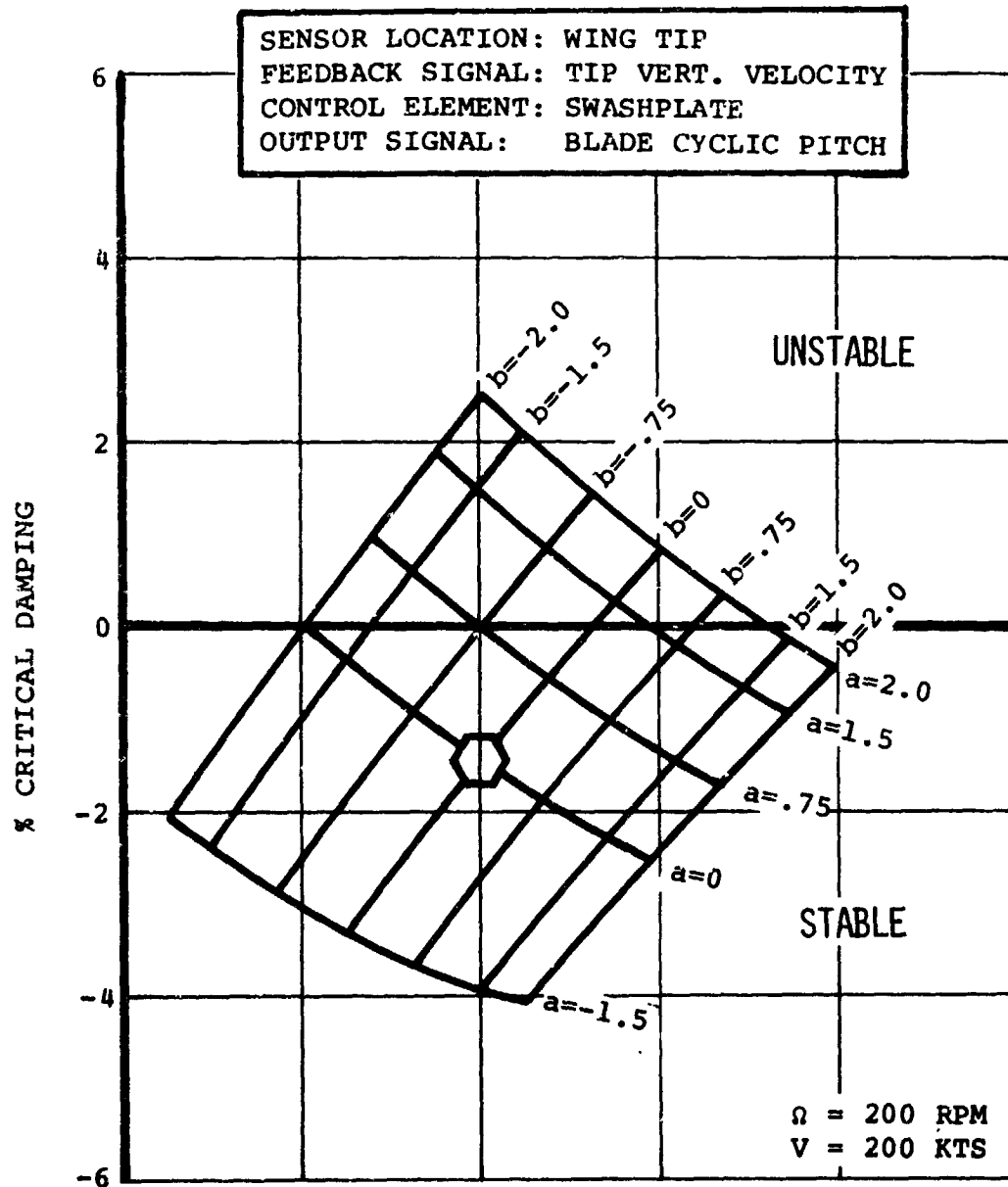
FIGURE 5.31 FEEDBACK EFFECT ON MODAL DAMPING OF
WING TORSION MODE

SENSOR LOCATION: WING TIP
FEEDBACK SIGNAL: TIP VERT. VELOCITY
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH
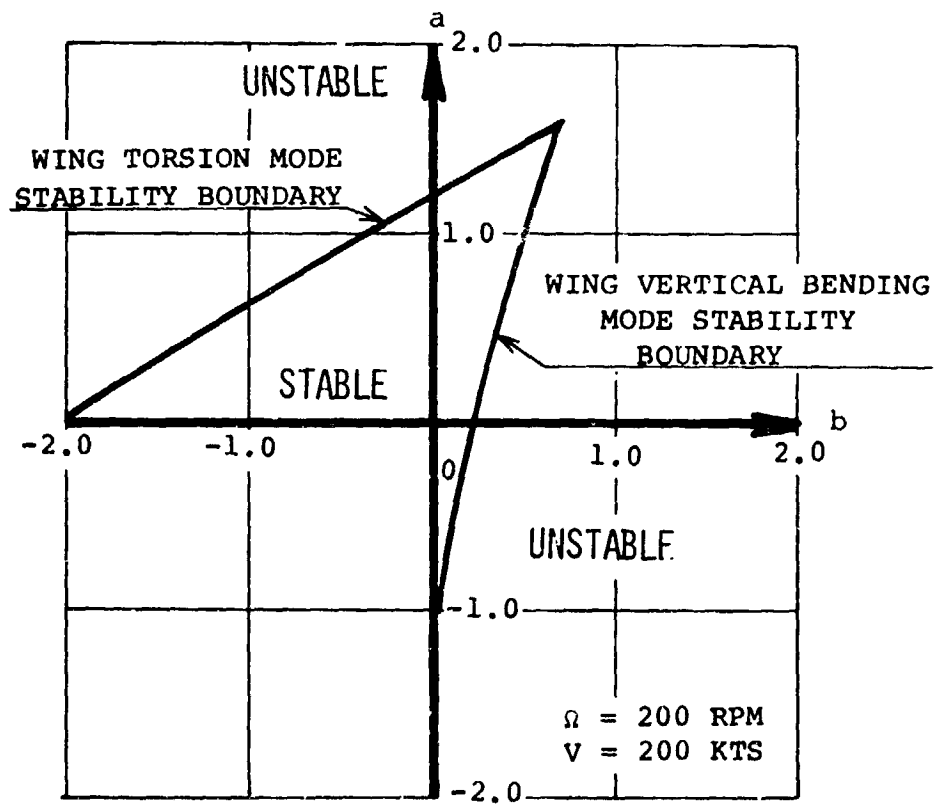
FIGURE 5.32 FEEDBACK EFFECT ON MODAL DAMPING OF AIRCRAFT
STABILITY FOR FEEDBACK COMBINATIONS

envelope (Figures 5.29 and 5.32) are a little larger, but exhibit the same general shape. A total gain envelope for the portions of the flight envelope investigated is shown in Figure 5.33. Assuming a time constant of .05 seconds and a desire to maintain a minimum of 1% critical damping in the torsion mode the optimum gain would be a = .4 and b = -.75 yielding a vertical bending mode damping of 11% at 150 knots.

From this exploratory study it was concluded that rate feedback showed great potential for adding damping to the wing vertical bending mode. There were several areas which needed further investigation:

    O   other feedback signals

    O   effect of actual hardware dynamics

    O   instabilities in other modes and corrective action.

The second portion of the study investigated these parameters using a refined optimization approach.

5.5 SYNTHESIS OF AN OPTIMIZED SYSTEM ACCOUNTING FOR HARDWARE COMPONENT DYNAMICS

Using the lessons learned from the initial studies the approach to the feedback system synthesis was refined and a more detailed study was undertaken. Specific additions to the analyses were

    O   refined approach to cyclic azimuth selection

    O   inclusion of actual hardware dynamics

    O   shaping and filtering of feedback signal.

SENSOR LOCATION: WING TIP
FEEDBACK SIGNAL: TIP VERTICAL VELOCITY
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH
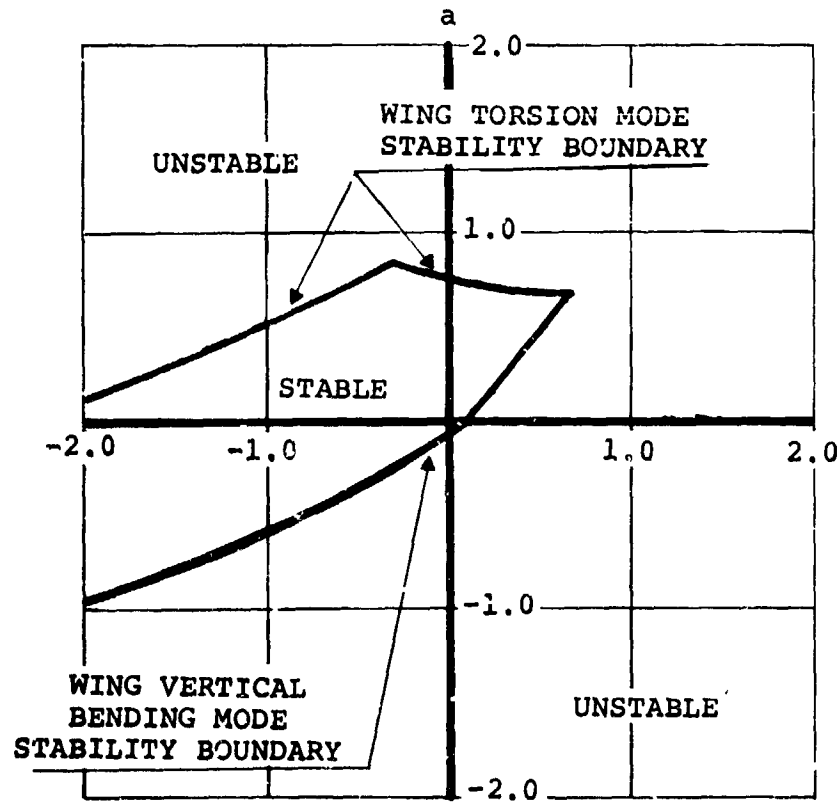
FIGURE 5.33.   COMPOSITE FEEDBACK GAIN ENVELOPE
FOR VARIOUS FLIGHT CONDITIONS

Instead of selecting the cyclic azimuth position on the basis
of the stability roots which reflect the magnitude and phase
of the control force, azimuth position was chosen on a magnitude
basis only and phase controlled if necessary by phase shaping.
Determination of the azimuth which yields maximum response in
wing vertical bending was accomplished by calculating the Bode
diagrams for a rosette of azimuth positions. Azimuth location
which yielded the largest response at the wing vertical bending
frequency was chosen as the feedback control setting.

Since system lags have a significant effect on the level of
damping achievable, it is clear that inclusion of actual system
hardware response characteristics would improve the quality of
any conclusions drawn about modal suppression feedback systems.
The model therefore used characteristics from the 26-foot rotor
set up tested in the 40x80-foot NASA-Ames tunnel. These include:

   O actuator dynamics

   O sensor transfer functions

   O swashplate control rod stiffnesses

Other pieces of hardware in the feedback loops had frequency
characteristics which would not affect operation of the feed-
back system (i.e., their break frequency was much higher than
any other frequency in the system).

The basic feedback sensor was an accelerometer mounted on the
nacelle in such a manner as to sense wing tip vertical accele-
ration. The position selected was nodal in the torsion mode.

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Acceleration, rate and position feedback were evaluated by shaping the accelerometer signals to determine if the intuitive choice of rate feedback was correct. Rate and position feedback signals were obtained by integrating the acceleration signal once and twice respectively (see Figure 5.2). The capability for filtering and phase shifting was made available to attenuate modes which might go unstable (e.g., the wing torsion mode) or shift the phase at the vertical bending frequency to get a pure position, rate or acceleration signal.

A rosette of cyclic azimuth positions were run for various flight conditions (rpm and velocity variations) and Bode diagrams plotted. From these the azimuth position (ψ_{rot}) was chosen which yielded the largest response in the wing vertical bending mode. It was ψ_{rot} = 60 degrees and is shown in Figure 5.34. Figures 5.35, 5.36, and 5.37 show the Bode plots for the acceleration signal for ψ_{rot}= 60 degrees. The plots for the remaining rpm and velocity conditions may be viewed in Boeing Document D160-10019-1, Pretest Calculation of Open Loop Frequency Response and Stability of High Rate and Low Rate Swashplate Feedback Systems of M222 26-Foot Rotor Test. The plots in this document have had a bandpass filter added to them. The filter is defined in the document.

With the chosen azimuth position several different systems were evaluated. For each system the hardware dynamics were included in the Bode plots and the gain-phase relationships evaluated to determine the need for phase shifting or filtering.

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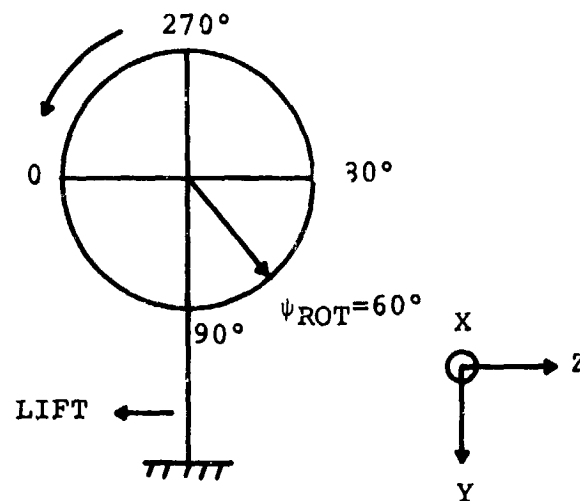
MATH MODEL SIGN CONVENTIONS

- $\Delta\theta = A_1 \cos \psi + B_1 \sin \psi$

    POSITIVE A_1 YIELDS NOSE DOWN PITCHING MOMENT (0° to 180°)
    POSITIVE B_1 YIELDS NOSE LEFT YAWING MOMENT (90° to 270°)

- Values of ψ_{ROT} demanded give magnitudes of A_1 and B_1 cyclic such that the maximum blade angle occurs at the azimuth position shown above.

FIGURE 5.34 MATH MODEL SIGN CONVENTIONS AND VECTOR ORIENTATION
FOR HI-RATE SYSTEM
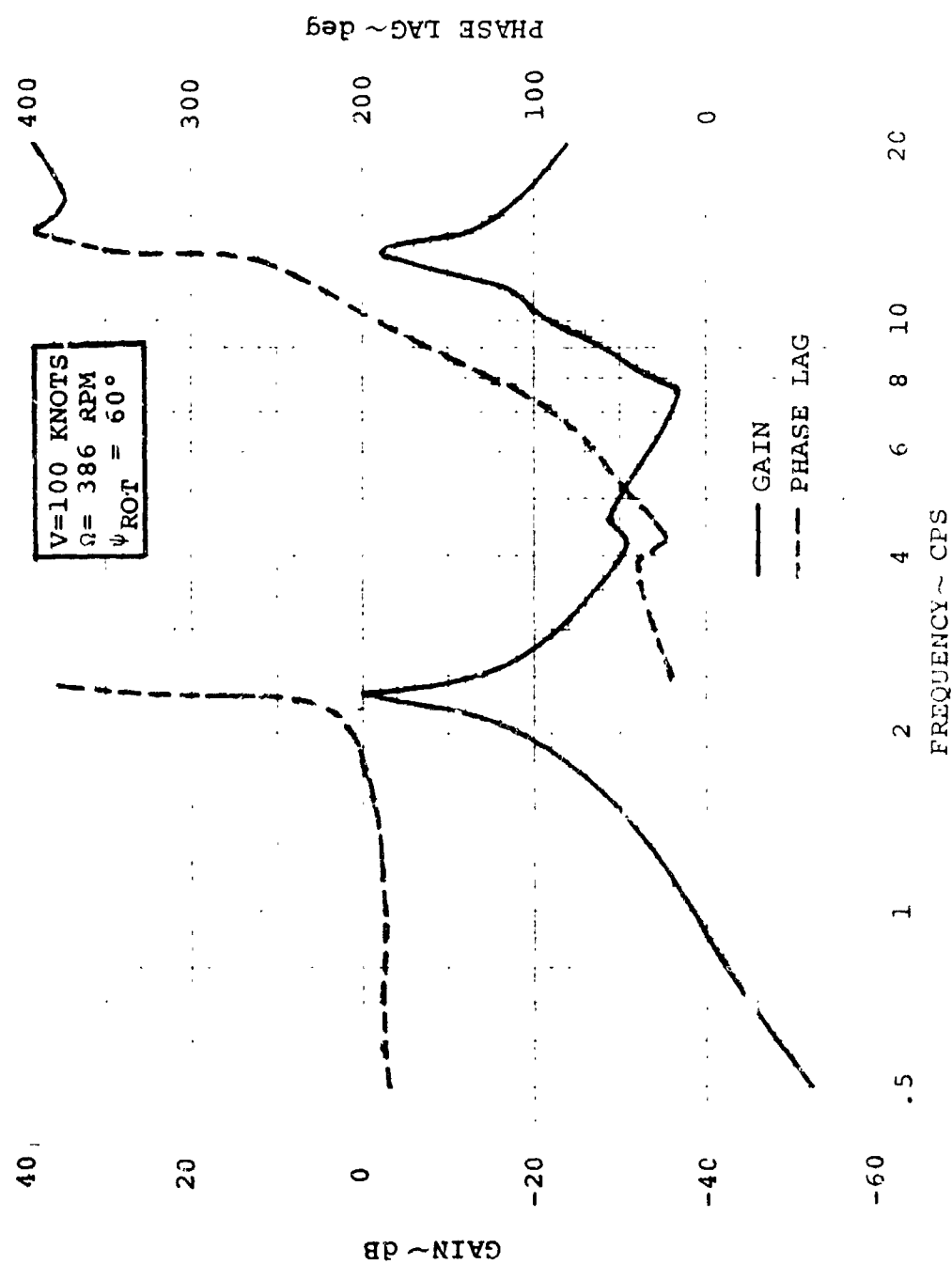
Figure 5.35. Aeroelastic Stability Augmentation System
Open Loop Frequency Response - Acceleration
Feedback - 100 Knot Cruise
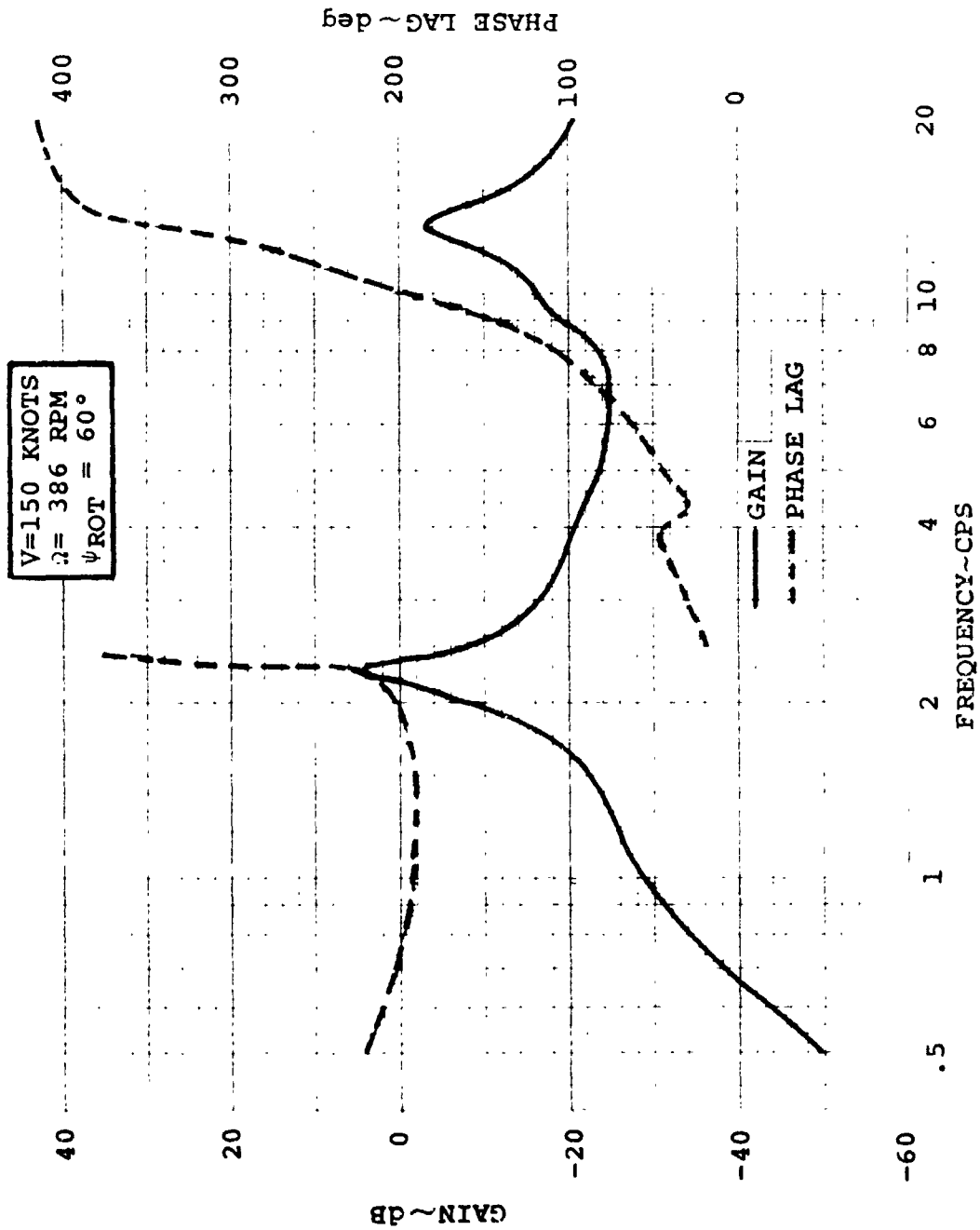
Figure 5.36. Aeroelastic Stability Augmentation System
Open Loop Frequency Response - Acceleration
Feedback - 150 Knot Cruise
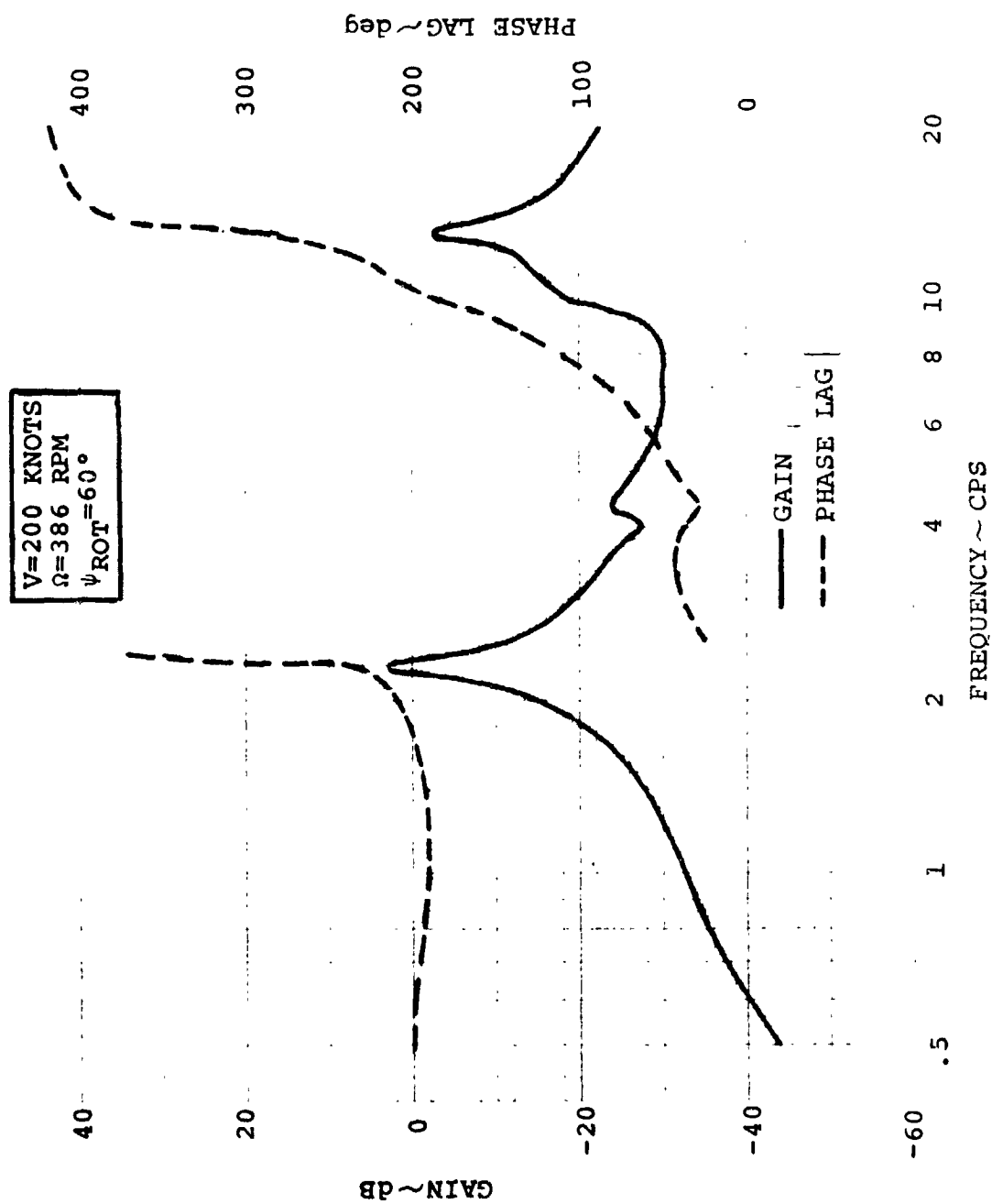
Figure 5.37. Aeroelastic Stability Augmentation System
Open Loop Frequency Response - Acceleration
Feedback - 200 Knot Cruise

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5.5.1 Acceleration Feedback

Acceleration feedback was the first to be evaluated because preliminary model testing had already indicated that useful improvement could be obtained with this system. On the block diagram of Figure 5.2, it is the loop without integrators. Figures 5.35, 5.36, and 5.37 shows the Bode plots for acceleration feedback at 386 RPM and 100 knots, 150 knots, and 200 knots. On a Bode plot instability is indicated when the phase is 180-degrees and the gain is 0 dB or greater; however, if the gain is to be increased the 180-degree crossing shoula be of concern. In each velocity case there is a 180-degree crossing at approximately 9 cps which for the mathematical model being evaluated is the wing torsion mode - the mode which went unstable in the simplified case. A fourth order narrow bandpass filter was put in the system to eliminate instabilities. Bode plots for the system with the filter are shown in Figures 5.38, 5.39,

and 5.40.   This takes care of stability at higher and lower frequencies.   The phase at 2.2 cps was not satisfactory and was shifted by 90-degrees using a phase shift network and the loop was closed. Stability root variation with gain is shown in Figures 5.41, 5.42, and 5.43. In the best case the wing

vertical bending damping was increased from 1.5% to 17%. This system was installed and tested on the 26-foot rotor in the Ames tunnel. Figures 5.44 and 5.45 show excellent correlation for the Bode analysis and the effect of gain on vertical bending damping. This correlation raises the confidence level in the program and leads us to believe that our physical representation of the feedback system is correct.
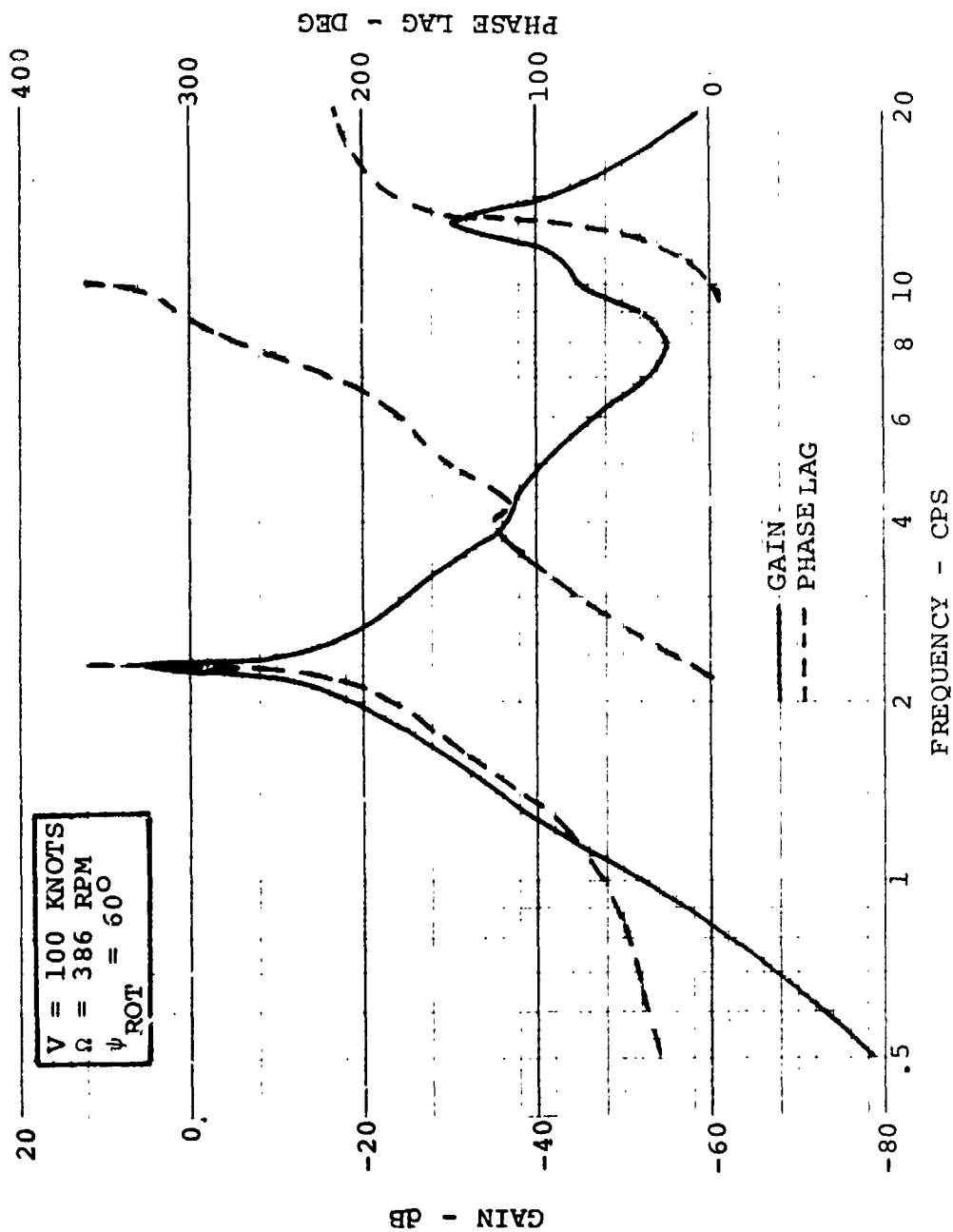
FIGURE 5.38. AEROELASTIC STABILITY AUGMENTATION
SYSTEM OPEN LOOP FREQUENCY RESPONSE
FILTERED ACCELERATION FEEDBACK - 100 KNOTS.
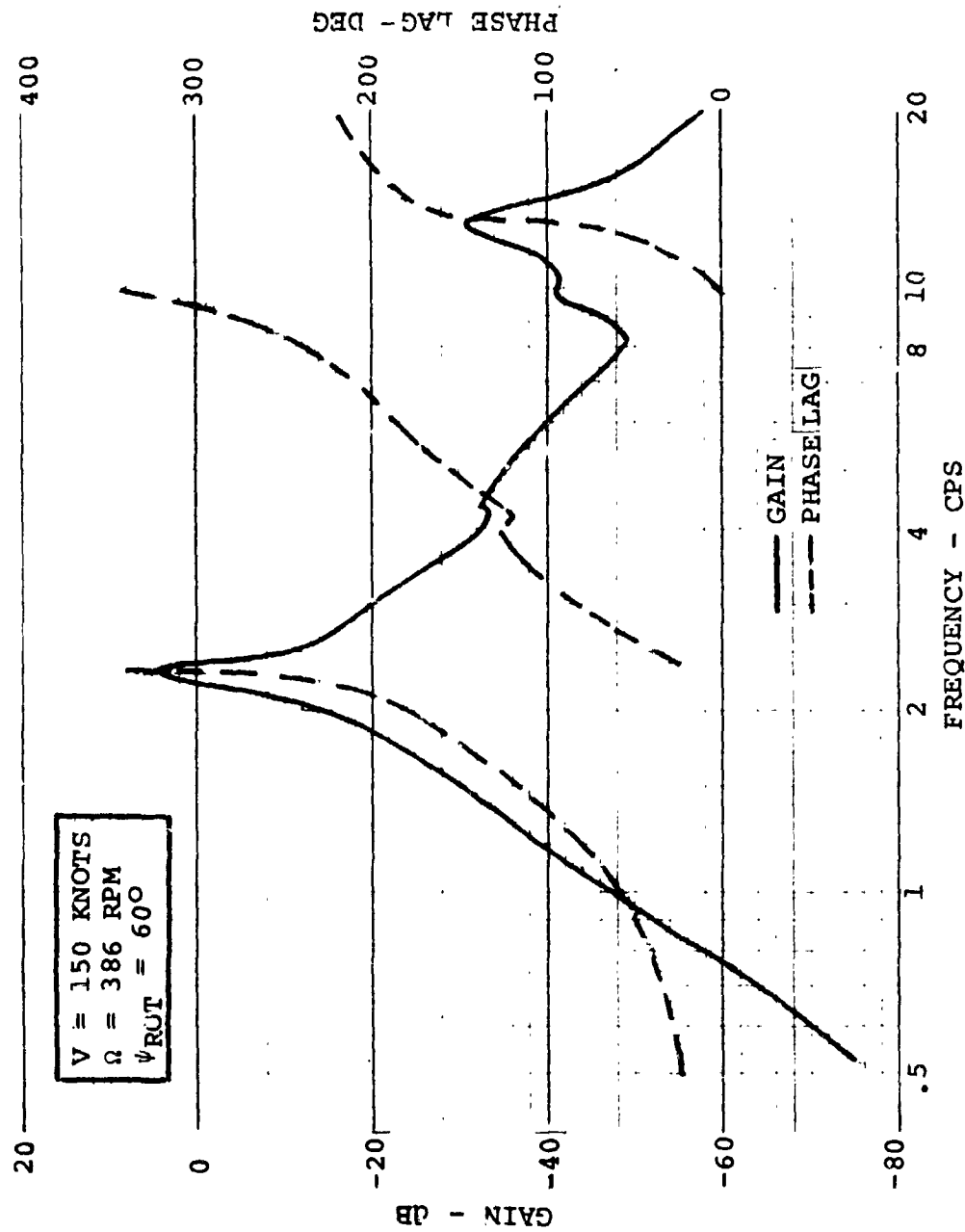
FIGURE 5.39. AEROELASTIC STABILITY AUGMENTATION
SYSTEM OPEN LOOP FREQUENCY RESPONSE
FILTERED ACCELERATION FEEDBACK - 150 KNOTS.
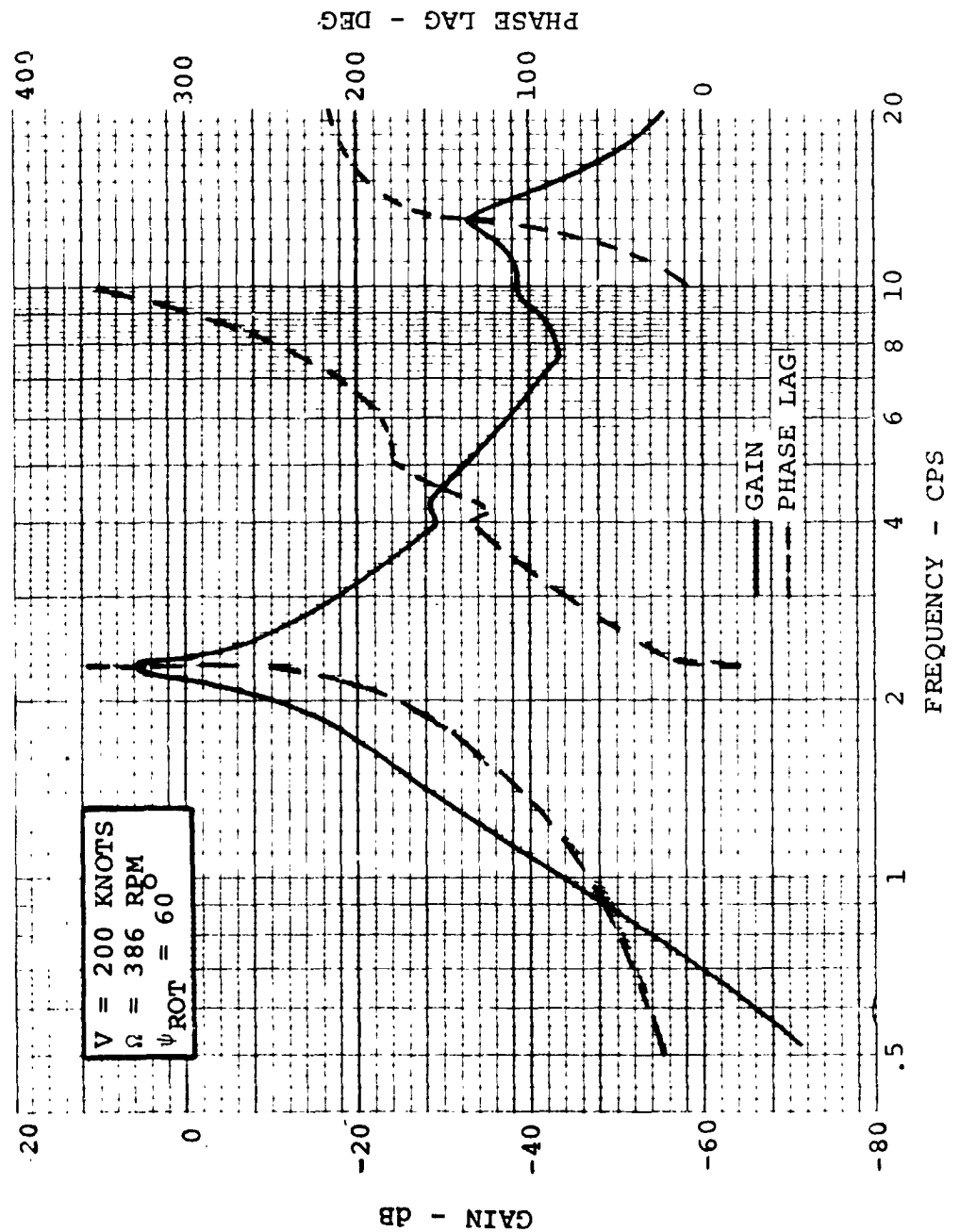
FIGURE 5.40. AEROELASTIC STABILITY AUGMENTATION SYSTEM
OPEN LOOP FREQUENCY RESPONSE FILTERED
ACCELERATION FEEDBACK - 200 KNOTS.

D222-10060-3
REV. A

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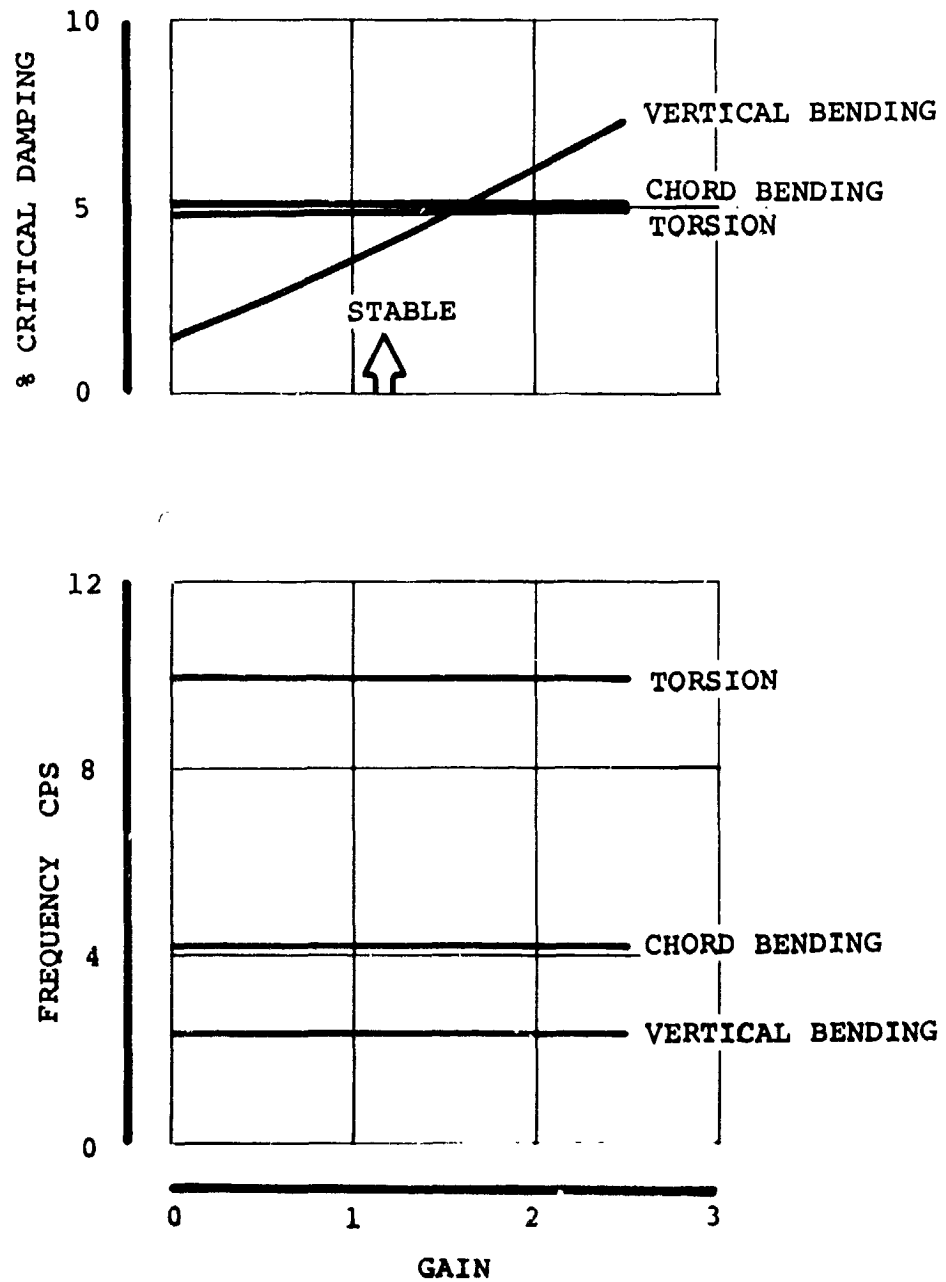
FIGURE 5.41.   EFFECT OF GAIN ON MODAL STABILITY ROOTS FOR
FILTERED ACCELERATION FEEDBACK - 100 KNOTS
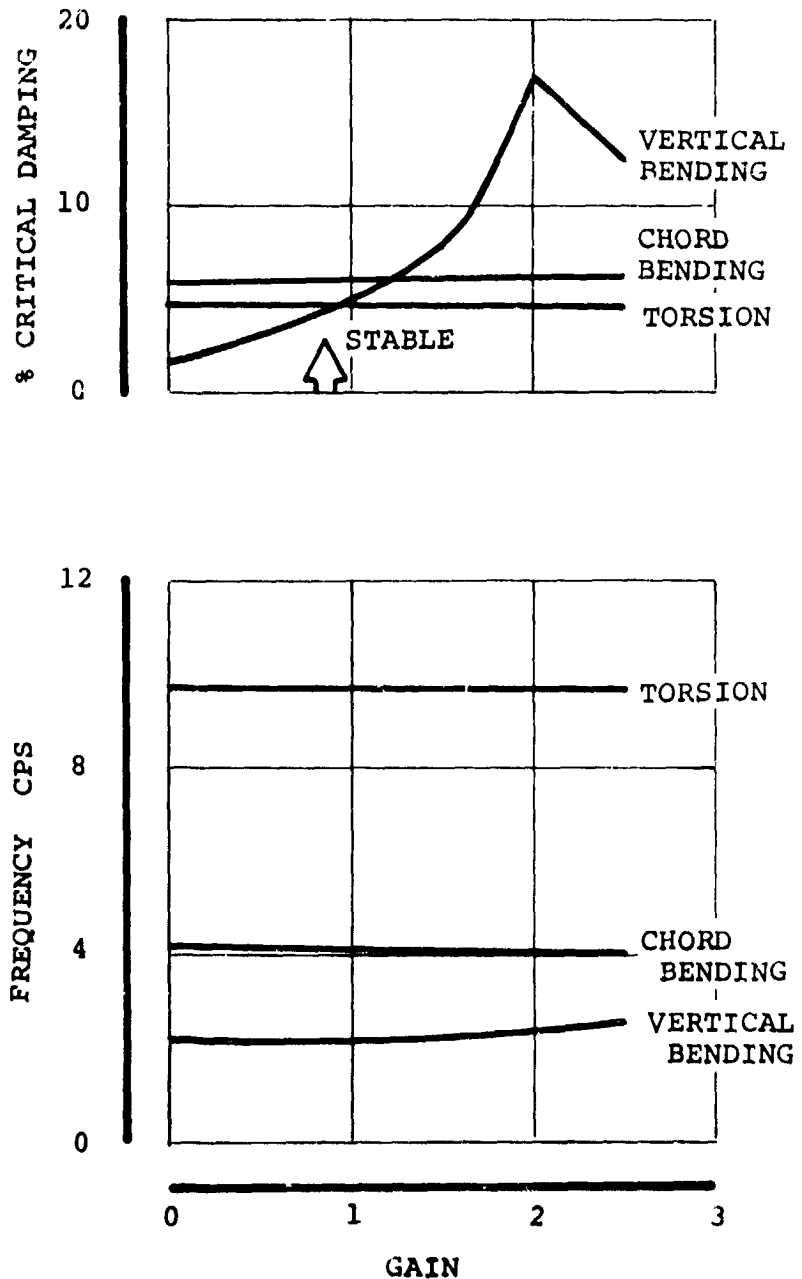
FIGURE 5.42. EFFECT OF GAIN ON MODAL STABILITY ROOTS FOR FILTERED ACCELERATION FEEDBACK - 150 KNOTS

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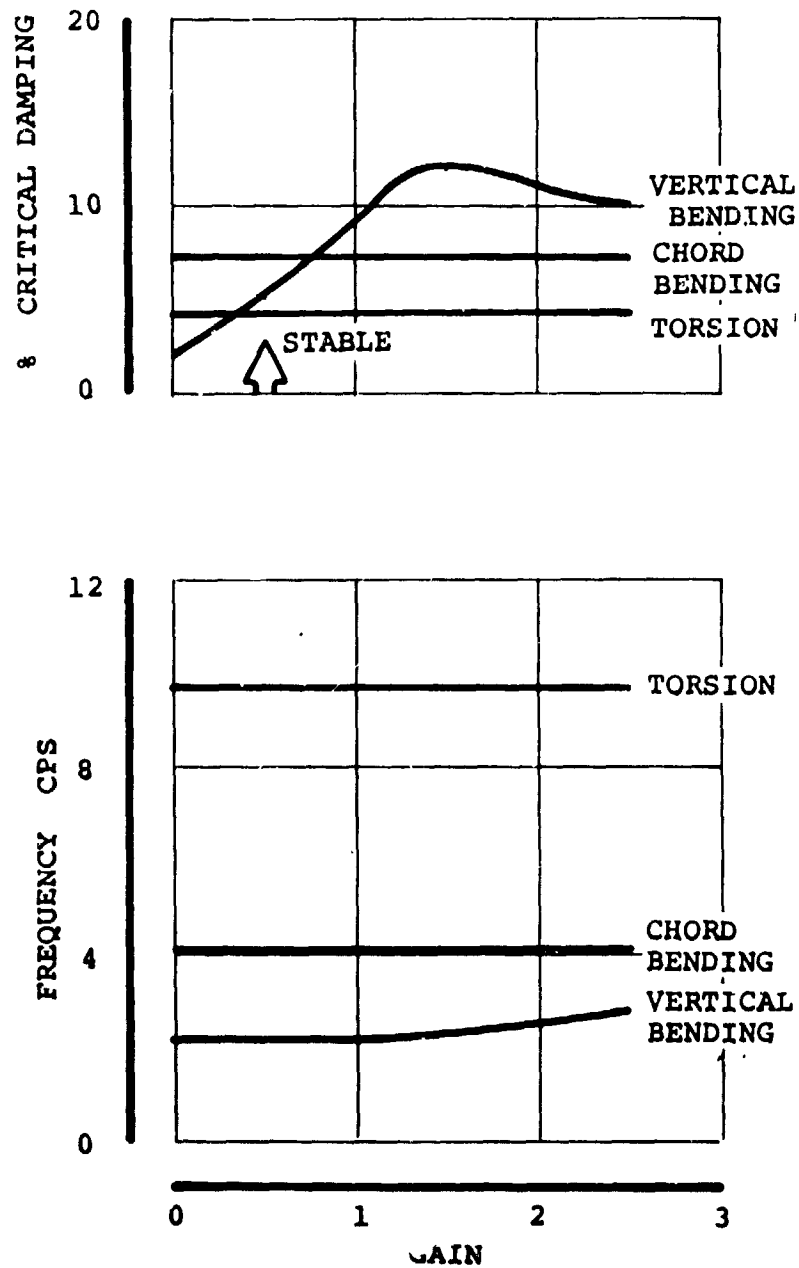
FIGURE 5.43.   EFFECT OF GAIN ON MODAL STABILITY ROOTS FOR
FILTERED ACCELERATION FEEDBACK - 200 KNOTS

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Post-test analysis of the system indicated that possibly use of acceleration feedback was not the most efficient method to increase stability, so rate and position feedback were evaluated analytically.

5.5.2 Rate Feedback

The second system evaluated was a rate feedback system. As seen in the Bodes of Figures 5.46, 5.47, and 5.48, the 180-degree crossing occurs when the gain is well down from the vertical bending peak at 2.25 cps. At test 40 dB separate the two. There are no filtering requirements to ensure stability. The only filter required may be on the accelerometer to remove excess noise. This filter will have a high break frequency and will not affect the system response. Stability roots were run for the system and appear on the root loci of Figures 5.49 and 5.50 for 100 and 200 knots. A damping ratio of .33 was achieved at a gain of .762 deg/ft/sec without degrading the other stability roots. It therefore appears that this system can be more effective than the acceleration system and permits simpler signal shaping.

5.5.3 Position or Deflection Feedback

The last feedback system to be evaluated was a position system. With this system it was hoped to raise the frequency of a given mode and thus postpone or eliminate a resonance condition.

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FIGURE 5.XX - HI-RATE OPEN LOOP FREQUENCY RESPONSE
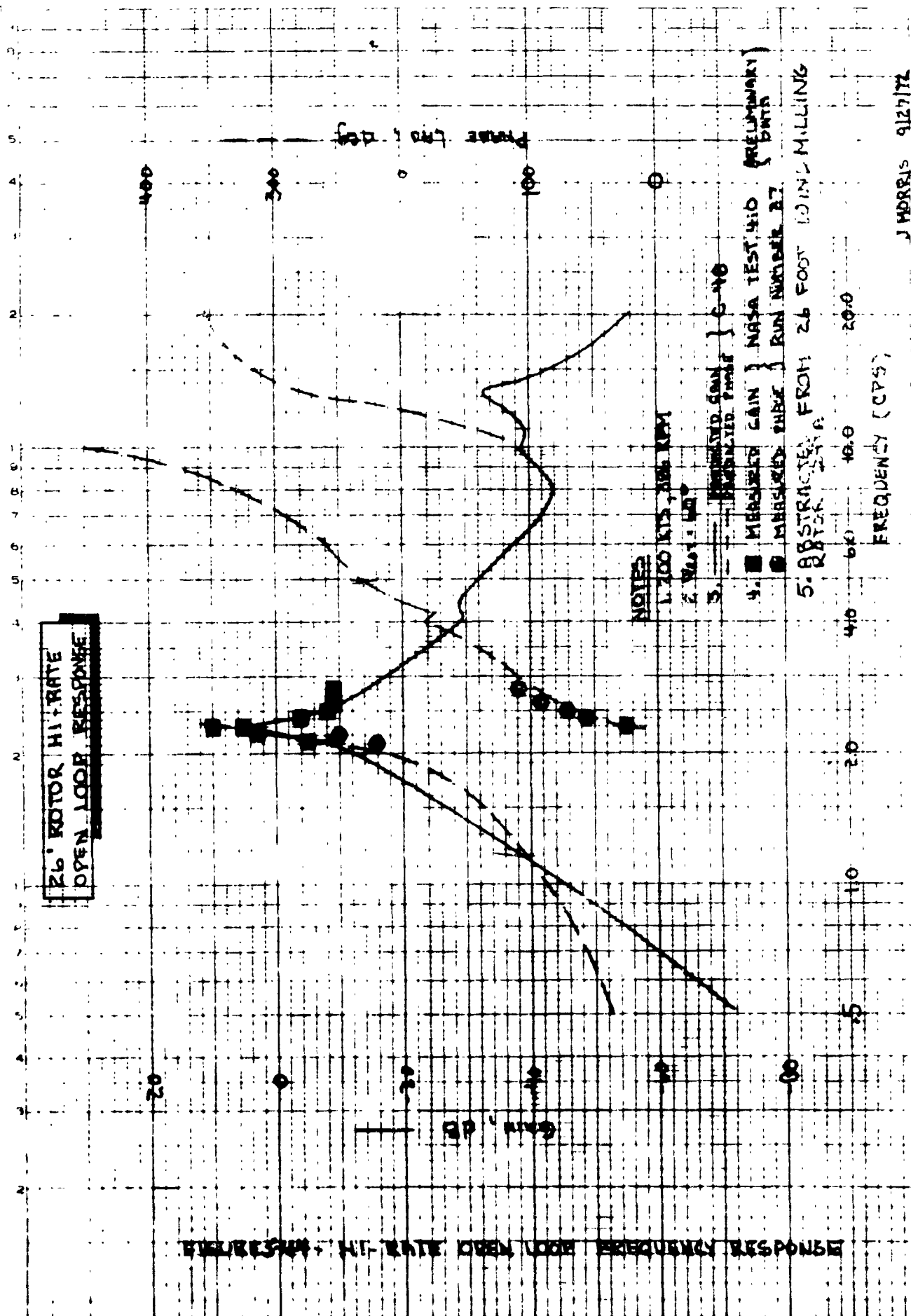
TEST 416: NASA AMES 40 X 80 FOOT TUNNEL
M-222 26' DIAMETER ROTOR TEST
NASA FULL STIFF WING

192 KTS 386 RPM
HIGH & LOW RATE LOOPS CLOSED
LOW RATE G_Y = 300

G_P = 204
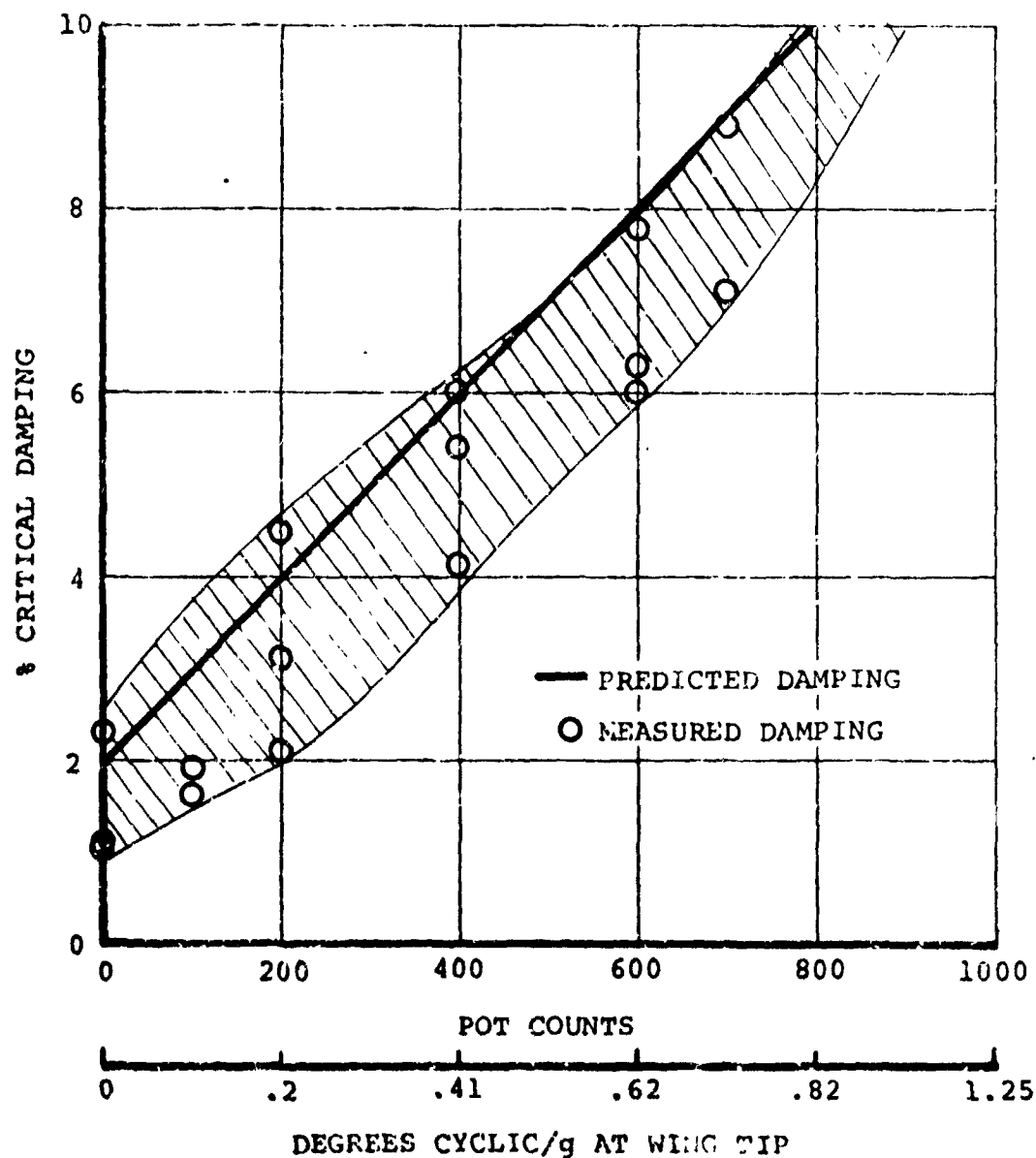
FIGURE 5.45. 26 FOOT ROTOR TEST DATA
DAMPING VERSUS HIGH RATE SYSTEM GAIN
AT 192 KNOTS, 386 RPM, WITH LOW RATE
SYSTEM ACTIVE
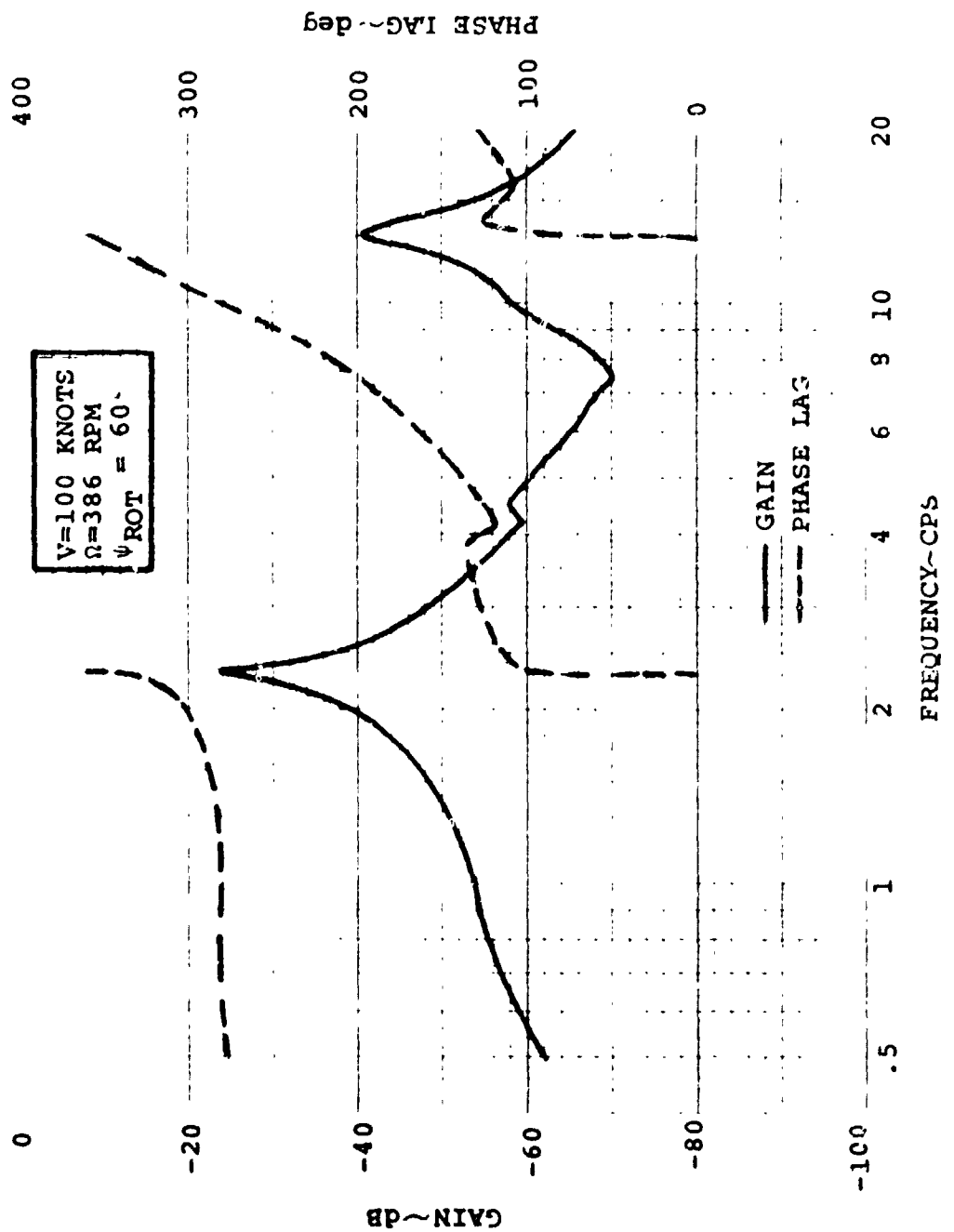
Figure 5.46. Aeroelastic Stability Augmentation System
Open Loop Frequency Response - Rate Feedback
- 100 Knots
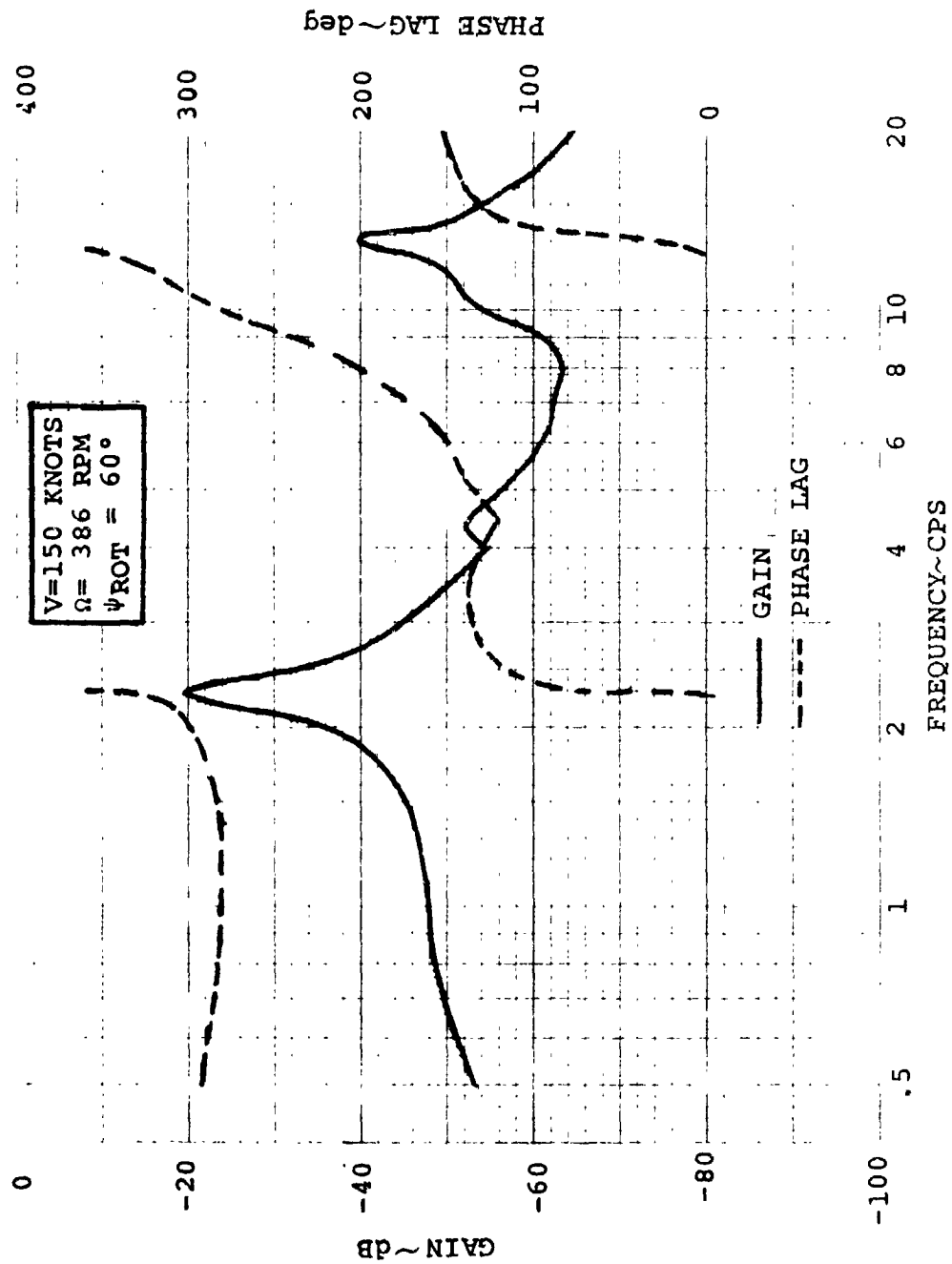
Figure 5.47. Aeroelastic Stability Augmentation System
Open Loop Frequency Response -
Rate Feedback - 150 Knots

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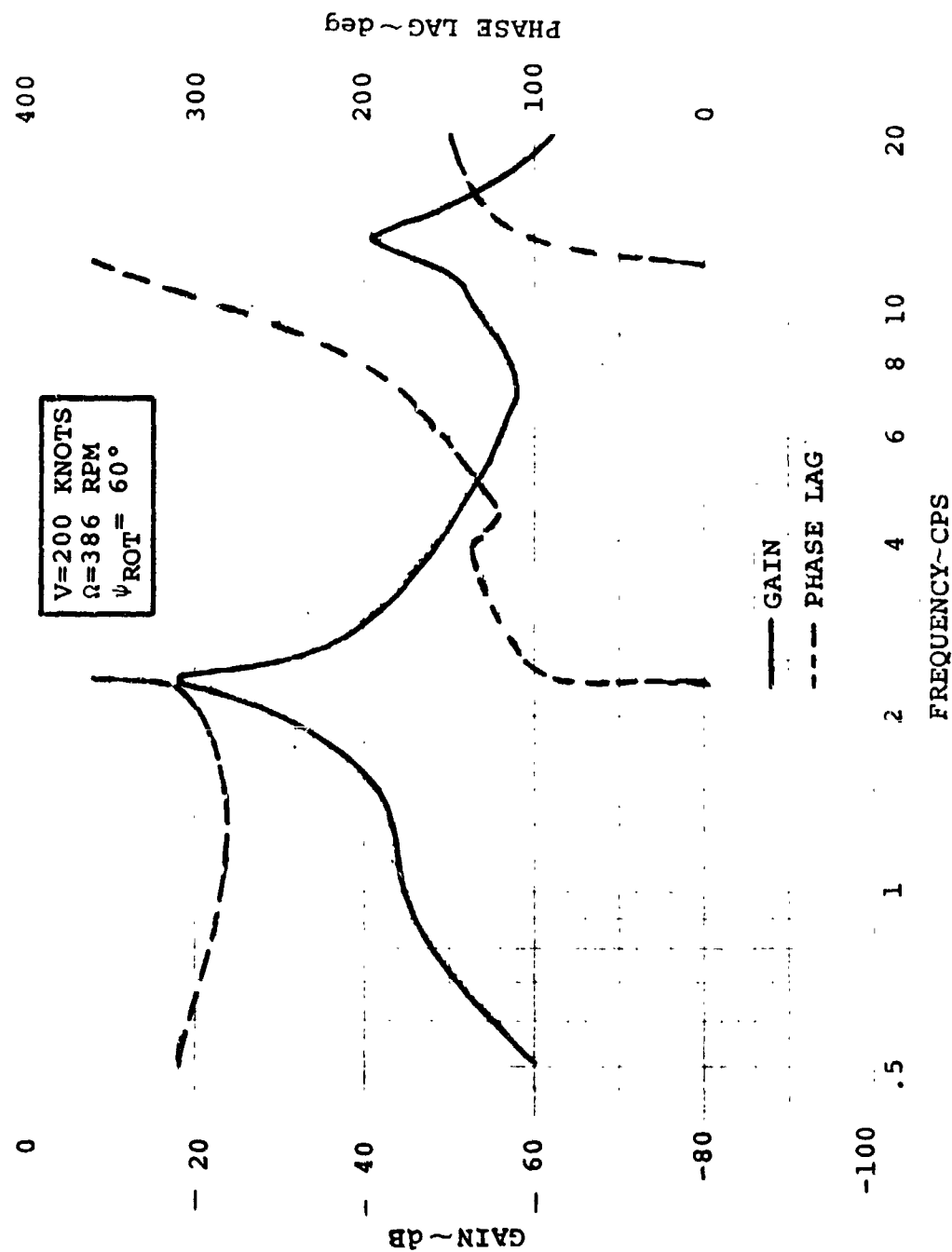
Figure 5.48. Aeroelastic Stability Augmentation System
Open Loop Frequency Response -
Rate Feedback - 200 Knots

CRUISE CONFIGURATION
386 RPM
100 KNOTS

GAIN~deg/ft/sec

WING VERTICAL
BENDING MODE

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$

DAMPING COEFFICIENT $(\zeta\omega_N)$, RAD/SEC
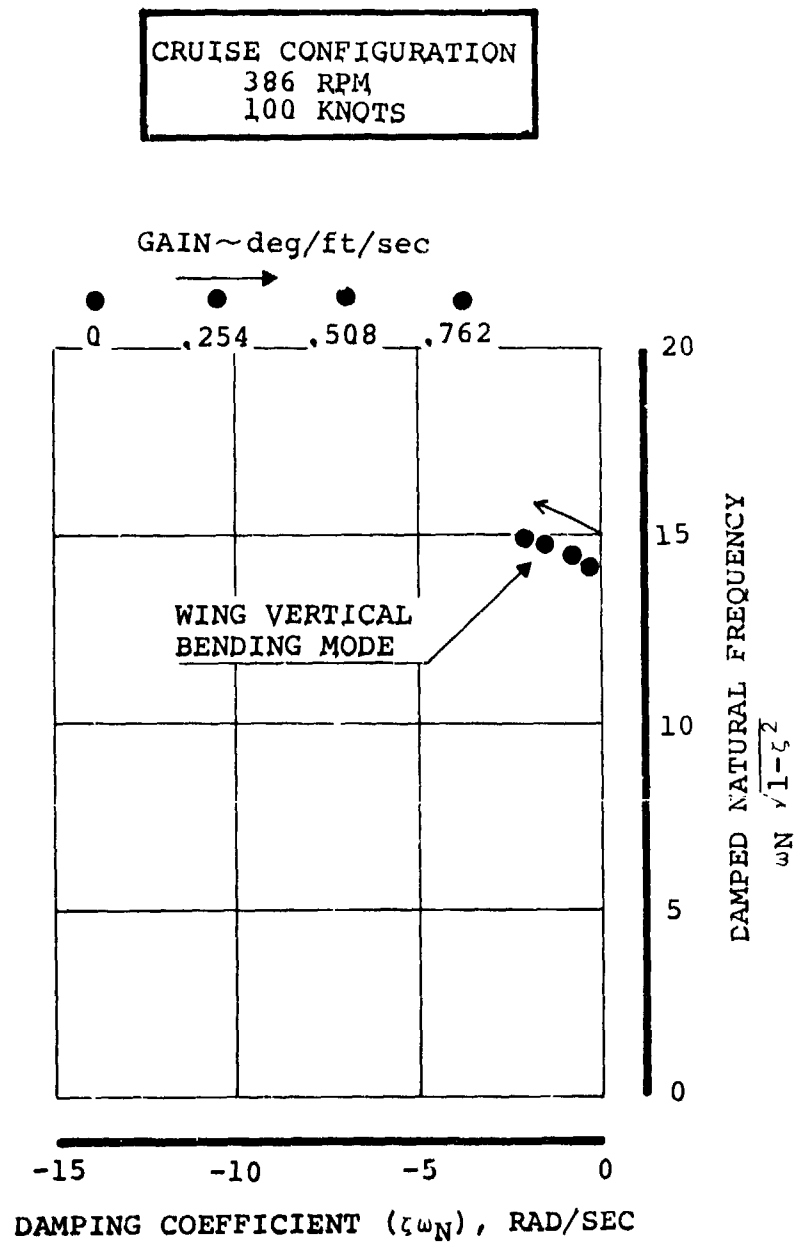
FIGURE 5.49. EFFECT OF RATE FEEDBACK ON WING
VERTICAL BENDING ROOTS
100 KNOTS

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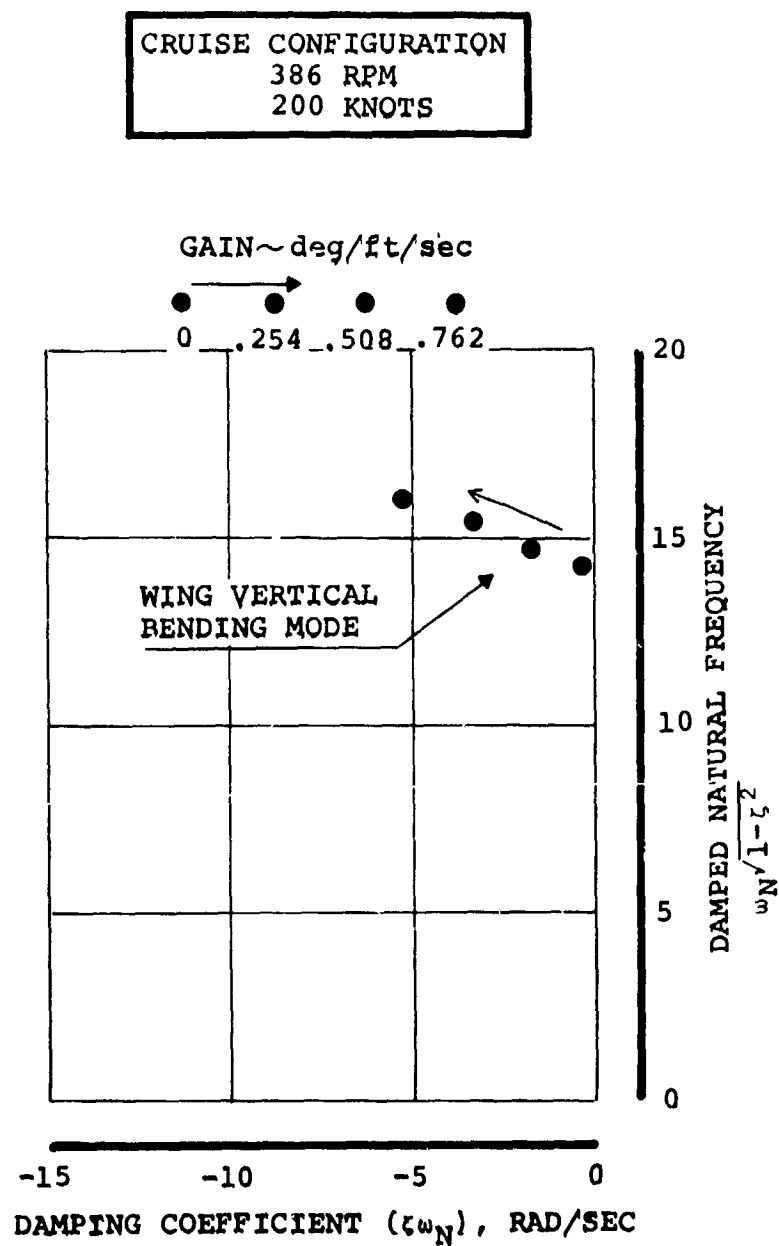
FIGURE 5.50. EFFECT OF RATE FEEDBACK GAIN ON WING
VERTICAL BENDING ROOTS 200 KNOTS

Consider the frequency spectrum of Figure 5.3. If the wing vertical bending frequency could be raised to 3 cps without degrading other modes the coalescence of frequencies known as air resonance could be postponed until approximately 400 knots. Then from a dynamics point of view the flight envelope has been widened. Evaluation of the system consisted of examining the Bode plots (Figures 5.51, 5.52, and 5.53) and calculating the stability roots. From the Bode plots there appear to be two 180-degree crossings which may cause trouble. The crossing at 13 cps does not seem to be troublesome because of the dB level being far enough down, however, the crossing at 2.3 cps is a problem and the vertical bending mode will go unstable at higher values of gain. Stability roots were run to see if the system went unstable before a desired level of frequency was reached (Figures 5.54 and 5.55). The root locus shows that high levels of gain are required and that the roots go unstable very quickly.

In summary it is seen that a system based on a rate signal is effective and less troublesome from the point of view of signal shaping than an acceleration based system. The position system could only be expected to be effective very near a stability boundary, so that for purposes of suppression of a basically stable but lowly damped mode, rate feedback should be chosen.

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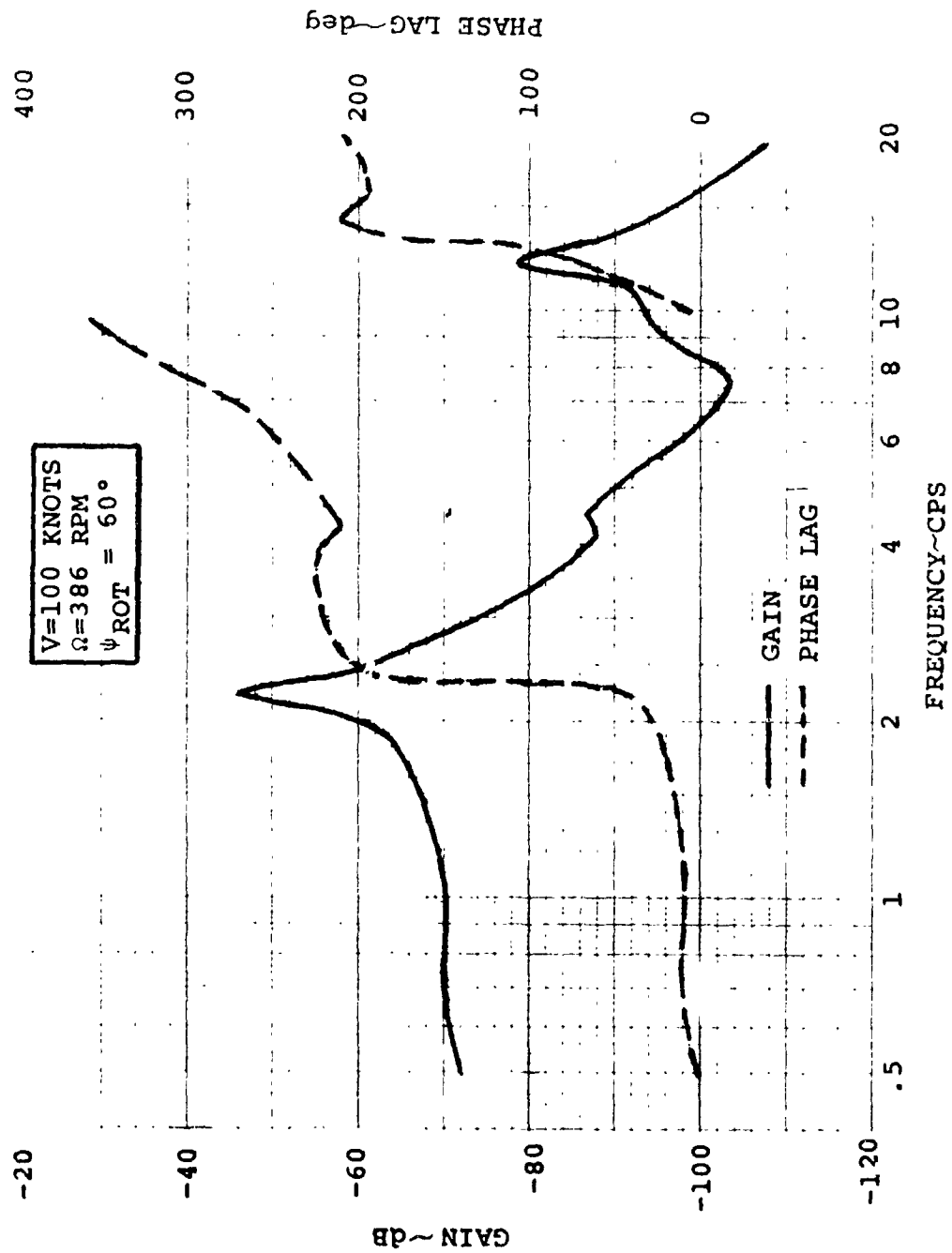
Figure 5.51. Aeroelastic Stability Augmentation System
Open Loop Frequency Response -
Position Feedback - 100 Knots
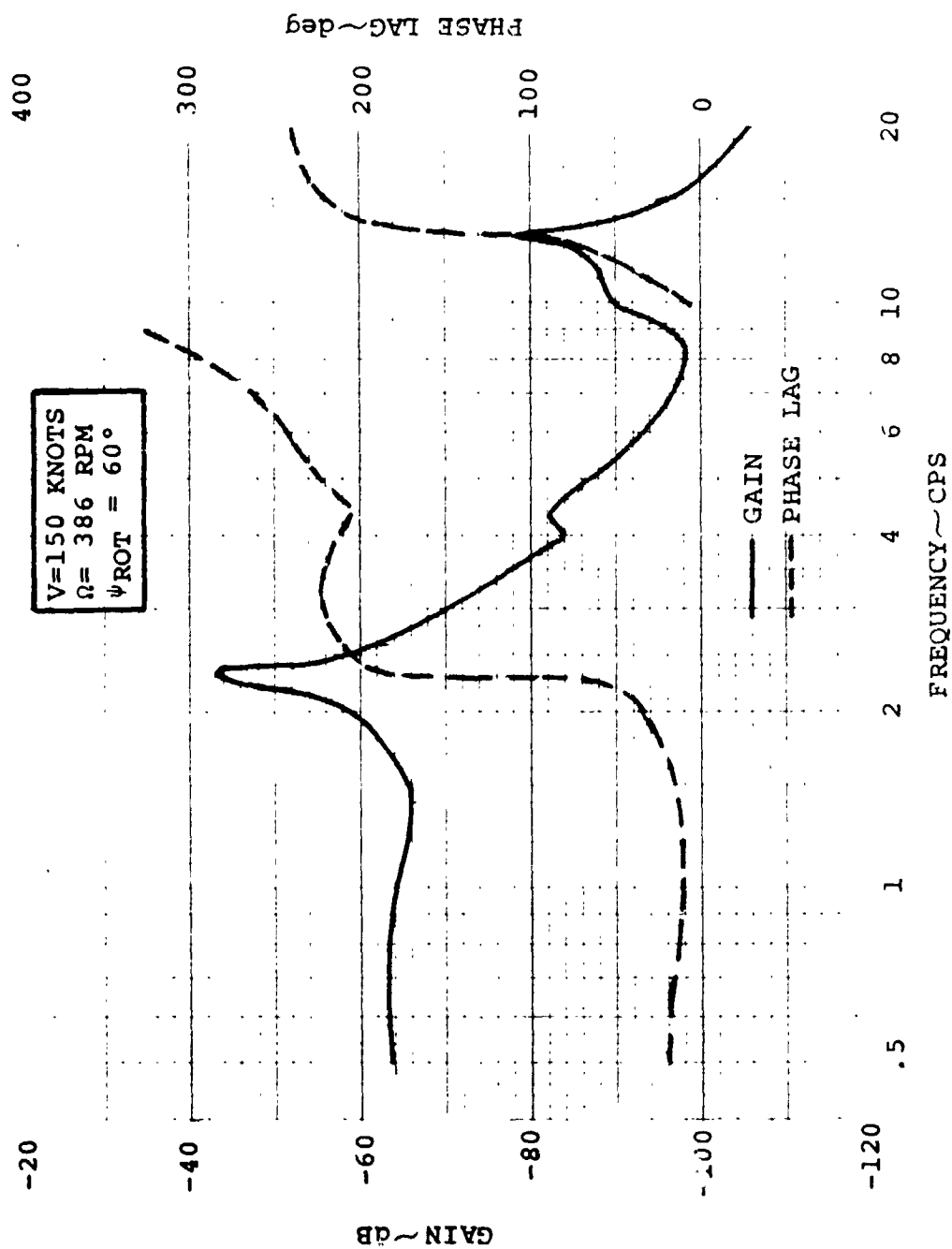
Figure 5.52. Aeroelastic Stability Augmentation System
Open Loop Frequency Response -
Position Feedback - 150 Knots
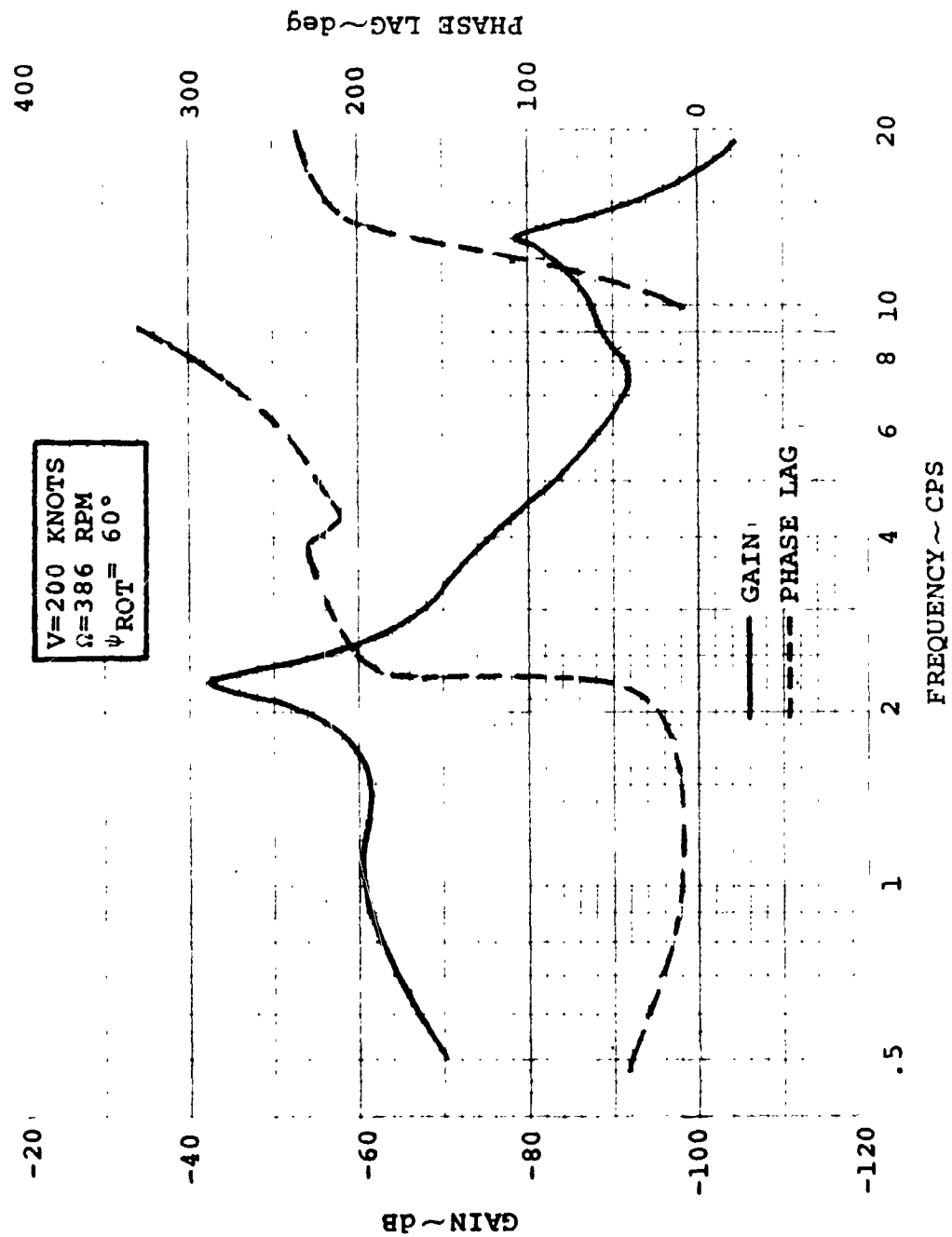
Figure 5.53. Aeroelastic Stability Augmentation System
Open Loop Frequency Response -
Position Feedback - 200 Knots

CRUISE CONFIGURATION
386 RPM
100 KNOTS
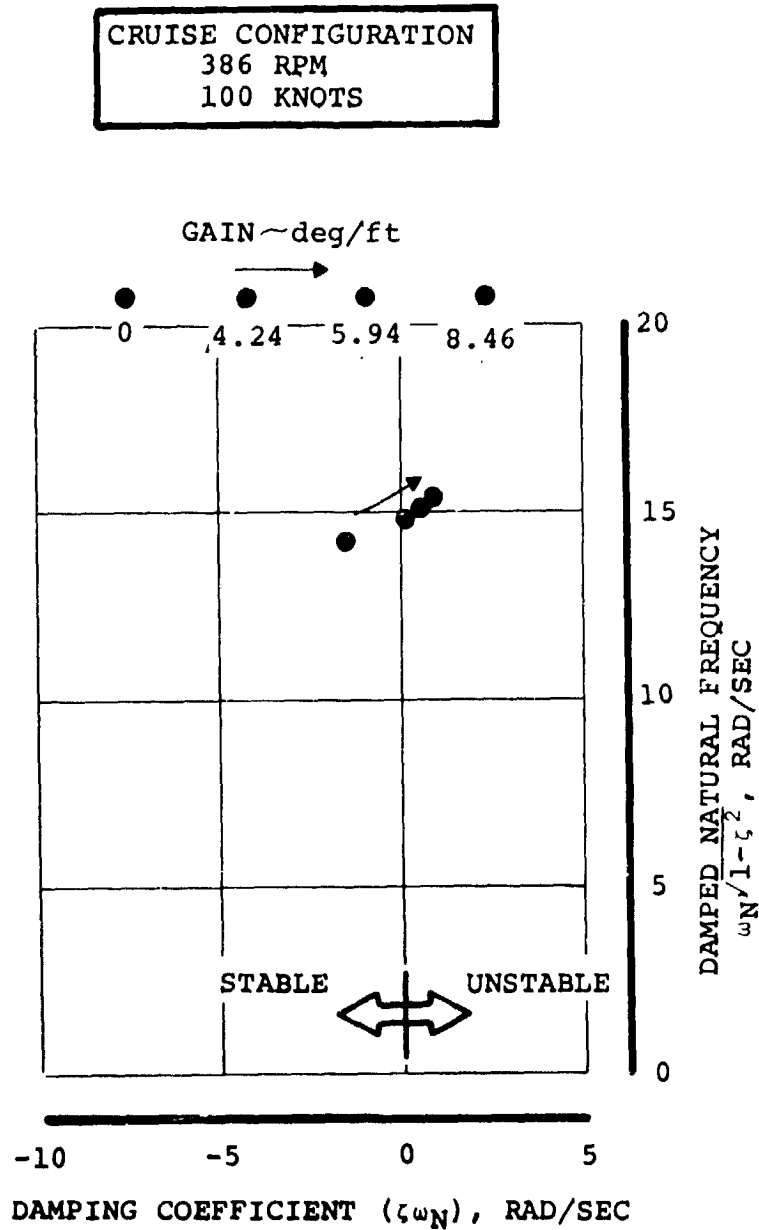
FIGURE 5.54:   EFFECT OF GAIN ON VERTICAL BENDING
STABILITY ROOTS - POSITION FEEDBACK -
100 knots
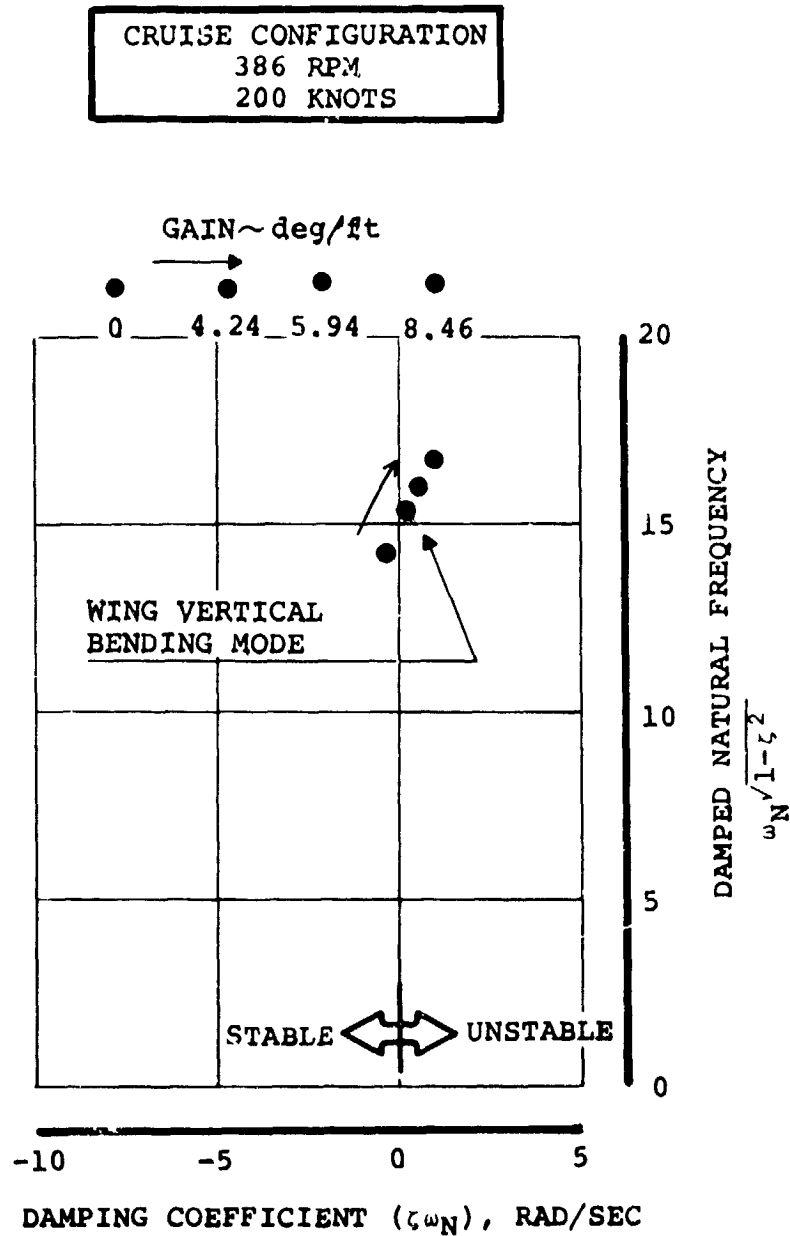
FIGURE 5.55. EFFECT OF GAIN ON VERTICAL BENDING STABILITY
ROOTS - POSITION FEEDBACK - 200 KNOTS

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5.6   PREDICTION FOR FURTHER DEVELOPMENT

The important difference between the exploratory analysis (section 5.4) and the more refined analysis of Section 5.5 lies in the fact that in the first approach the azimuth is used to control the system phase relationships, while in the second, force level rather than phase is the criterion for azimuth selection. Hence, in terms of effectiveness of the system per degree of cyclic control authority, the second approach is better: the first system is compromised from the start since some of the authority is being used to control phase instead of generating force.

Following this line of thought it is clear that a still better system may be defined if the available authority is used to generate force only in the direction which is effective. To accomplish this the concept of swashplate "wobble" or phasing between the A_1 and B_1 feedback signals must be introduced.

5.6.1   Phase Swashplate Feedback

Under static conditions the direction of hub force and moment may be controlled completely by selection of the proper ratio A_1 and B_1. This ratio may be, and frequently is, expressed as a unique axis direction or azimuth about which the swashplate tilts. When dynamic situations are considered it is highly restrictive to continue to think in terms of azimuth since it imposes the constraint of zero phasing between the A_1 and B_1 signals.

The question then arises whether the introduction of relative phasing between the A_1 and B_1 signals can be beneficial in a dynamic system such as those for modal suppression discussed above. To address this question the behavior of the hub forces

resulting from a swashplate oscillation will be discussed more in detail.

It is now known from analysis and test that when the swashplate is given a pitch or yaw oscillation of constant amplitude the resulting oscillatory normal force and side force are phased relative to the swashplate motion and to each other. This means that there is no combination of A_1 and B_1 which can be represented by an azimuth angle which will provide normal force only. Hence, unless special steps are taken some proportion of the authority is dissipated in side force. The special steps involve phasing the B_1 and A_1 feedback signals with respect to each other as derived below.

Stating this in a convenient mathematical form

Normal force due to $A_1 \propto A_1(1 + i \tan \phi_1)$

Side force due to $A_1 \propto A_1(1 + i \tan \phi_2)$

Similarly

Normal force due to $B_1 \propto B_1(1 + i \tan \phi_2)$

Side force due to $B_1 \propto B_1(1 + i \tan \phi_1)$

Where A_1, B_1 are amplitude of pitch and yaw cyclic oscillations

ϕ_1 is phase of NORMAL FORCE WITH RESPECT TO A_1

ϕ_2 is phase of SIDE FORCE WITH RESPECT TO A_1

The net side force is given by

$F_Y \propto A_1(1 + i \tan \phi_2) + B_1(1 + i \tan \phi_1)$

For $F_Y = 0$

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$$\frac{A_1}{B_1} = -\frac{1 + i \ Tan \ \phi_1}{1 + i \ Tan \ \phi_2} \ .$$

Hence the condition of $F_Y = 0$ is fulfilled when the relative amplitude

$$\frac{A_1}{B_1} = -\frac{1 + Tan^2 \ \phi_1}{1 + Tan^2 \ \phi_2} \ .$$

And the phase of A relative to B i

$$\theta = Tan^{-1} \ \frac{Tan \ \phi_2 - Tan \ \phi_1}{1 + Tan \ \phi_1 \ Tan \ \phi_2}$$

$$= (\phi_2 - \phi_1)$$

This means that a swashplate motion is identified which does not produce any side force, but generates additional normal force. That is, if normal force and side force produced by a swashplate oscillation have a phase angle θ with respect to each other, then imposing this same phase angle on the A_1 and B_1 swashplate motions fed back will effectively reduce the side force to zero. An investigation of the additional effectiveness released by utilizing the swashplate motion to produce only normal force was not possible within the scope of the present contract, but it is clear from the above analysis that determination of this factor is a matter of tedious but elementary manipulation.

In physical terms this means that the feedback system should be designed to produce a wobble in the swashplate rather than a rotation about one unique azimuth axis. In terms of hard-

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ware, the only requirement is that independent phase

shifting networks be provided for the A_1 and B_1 channels

instead of a single device as envisioned in the earlier

discussions.

5.6.2 Advantages of Phased or Wobbling Swashplate Control

Immediately identified advantages of phased swashplate control

are as follows:

O   The fullest use is made of the permitted authority
    of the system.

O   Unwanted side forces are eliminated. In the studies
    and tests made to date these have not proved to be a
    problem; however, it is not difficult to envision
    a situation where side force would stimulate nacelle
    yaw and possibly introduce new instabilities. Gener-
    ating only the force specific to the particular mode
    whose damping is to be increased removes this
    potential problem.

5.7 CONCLUSIONS

A number of different approaches are possible in designing a

swashplate feedback system for modal suppression or damping

augmentation. The differences lie in the control of phase

and direction of the forces generated by the system. At the

simplest level,the best azimuth angle for the swashplate is

found with no special account taken of the direction of forces

generated. At the second level,the swashplate angle is

selected to produce maximum force level in the direction of the mode, and the phase angle is then controlled electronically. In the most efficient system which eliminates potential side effects and makes optimal use of system authority, the swashplate is given a controlled precession or wobble by introducing a phase angle between the pitch and yaw commands. The difference in cost between the three systems is minimal since only signal conditioning is involved; hence, any practical application of feedback for modal ' opression should use the phased swashplate control principle.

6. BLADE TRANSIENT LOADS IN HOVER

6.1 BACKGROUND AND OBJECTIVES

The blade loading discussed in Section 3 arises from skewed
flow conditions, and may be alleviated by the application of
cyclic pitch in appropriate amounts. In hover, a different
type of loading is caused by the rise of cyclic pitch to
control the aircraft: this produces coriolis blade loads in
the steady state and transient blade response as a result of
rapid application of cyclic. There is at present no indi-
cation that blade transients generated by cyclic are a
problem, as borne out by experience on hingeless rotor heli-
copters such as the BO-105. Nevertheless, it is of interest
to know whether response in the blade modes may be attenuated
by the use of feedback control through the swashplate, as an
alternative to the more commonly used δ_3 and α_2 types of
mechanical feedback coupling. This type of coupling derives
its effectiveness by changing the blade frequency character-
istics and would not be admissible for use in a hingeless
rotor of optimized design.

The principal restriction of kinematical coupling is that the
change in blade angle proportional to blade flap or lag are
also in phase with the variable to be modified. Swashplate
feedback is not so restricted since rates of blade flap or
lag may be sensed and this type of signal, when fed back, is
equivalent to the addition of damping. The ability to augment

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damping in the blade modes without significantly affecting
the frequency characteristics would be an advance in the
state-of-the-art which might prove valuable in the future.

To examine the feasibility of such an accomplishment a
limited exploratory study was made to determine if feedback
would be an efficient method to increase the damping in the
lead-lag mode. The study was conducted in two stages:

o system sensing ideal signal, i.e., blade
   lead-lag rate

o system sensing hub in-plane velocity.

Although the investigation did not delve deeply into signal
shaping and the optimum configuration, the conclusions drawn
are encouraging and indicate that should the requirement arise,
a practical system to augment blade damping can be designed
using cyclic feedback.

6.2 MATHEMATICAL MODEL

The mathematical model used was representative of the symmetrical
degrees of freedom of a tilt rotor aircraft in hover. Blade
flap and lag modes as well as wing vertical bending, chord,
and torsion were included in the model. Representation of
the wing elastic modes and the rotor characteristics is
similar to the Model 222 tilt rotor aircraft.

Feedback loop characteristics have been represented by
transfer functions of control system hardware which have been
determined theoretically or by test.

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6.3 EVALUATION OF SYSTEM SENSING IDEAL SIGNAL

In order to determine a maximum boundary of damping that
could be put into the lead-lag mode, the rate of the mode
itself was fed back into the cyclic command. It was
recognized that this was hardly the basis for a practical
system, nevertheless, the results of this study establish
criteria for more practical systems. From the block diagram
of Figure 6.1, one can see the operation of the system. Blade
lead-lag velocity is sensed and filtered through a low-pass
filter with a break frequency at 2 cps. The filter was
chosen so that signals of higher frequency than $(\Omega-\omega_L)$ would

be attenuated. The filtered cyclic was then fed back through
the cyclic actuators in such a manner as to have the maximum
cyclic blade angle occur at either 45-degrees or 225-degrees.
These two azimuths were chosen to yield a cursory idea of the
proper quadrant in which maximum blade angle should occur.
Results of the study may be reviewed on Figures 6.2, 6.3, and
6.4. Figure 6.2 indicates that an azimuth of 225-degrees is
the proper choice to increase damping in the $(\Omega-\omega_L)$ mode.

Over the gain range investigated, the damping was increased
from 2% critical to 13% critical - a significant improvement
in terms of structural damping. Root migration with gain
is illustrated in Figure 6.3. Note that the path of the
roots indicates that pure rate feedback is not being obtained
(pure rate feedback is characterized by a root migration that

190
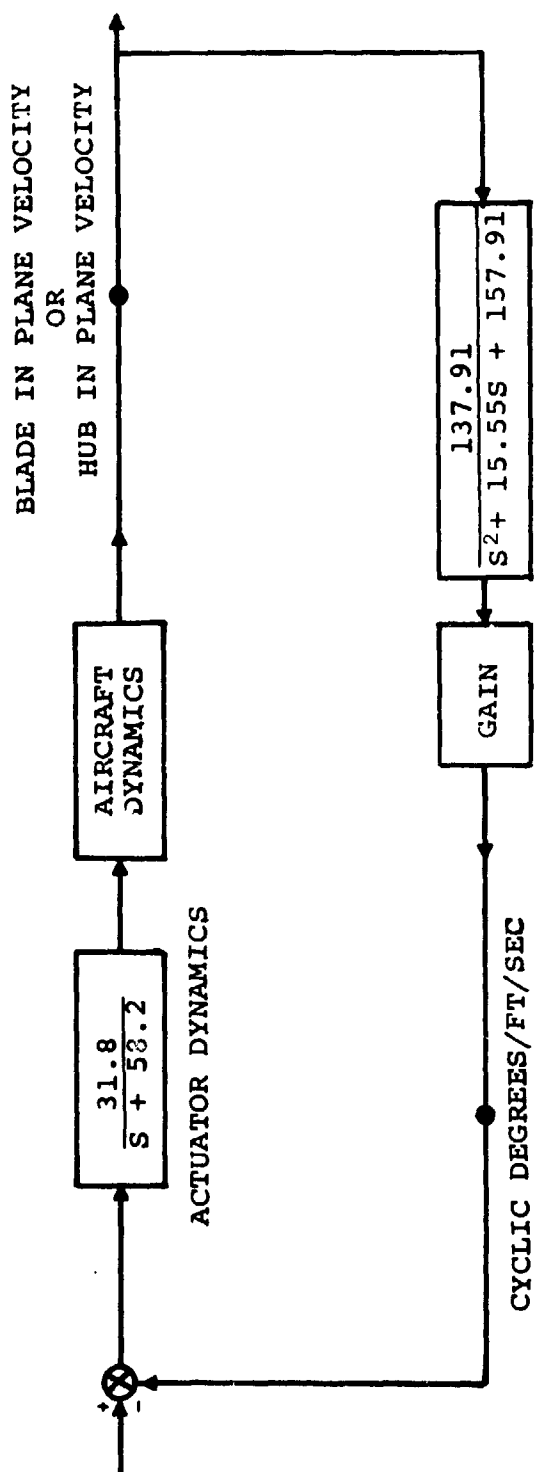
FIGURE 6.1. LEAD-LAG FEEDBACK BLOCK DIAGRAM

SENSOR LOCATION: BLADE TIP
FEEDBACK SIGNAL: BLADE LEAD-LAG RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH
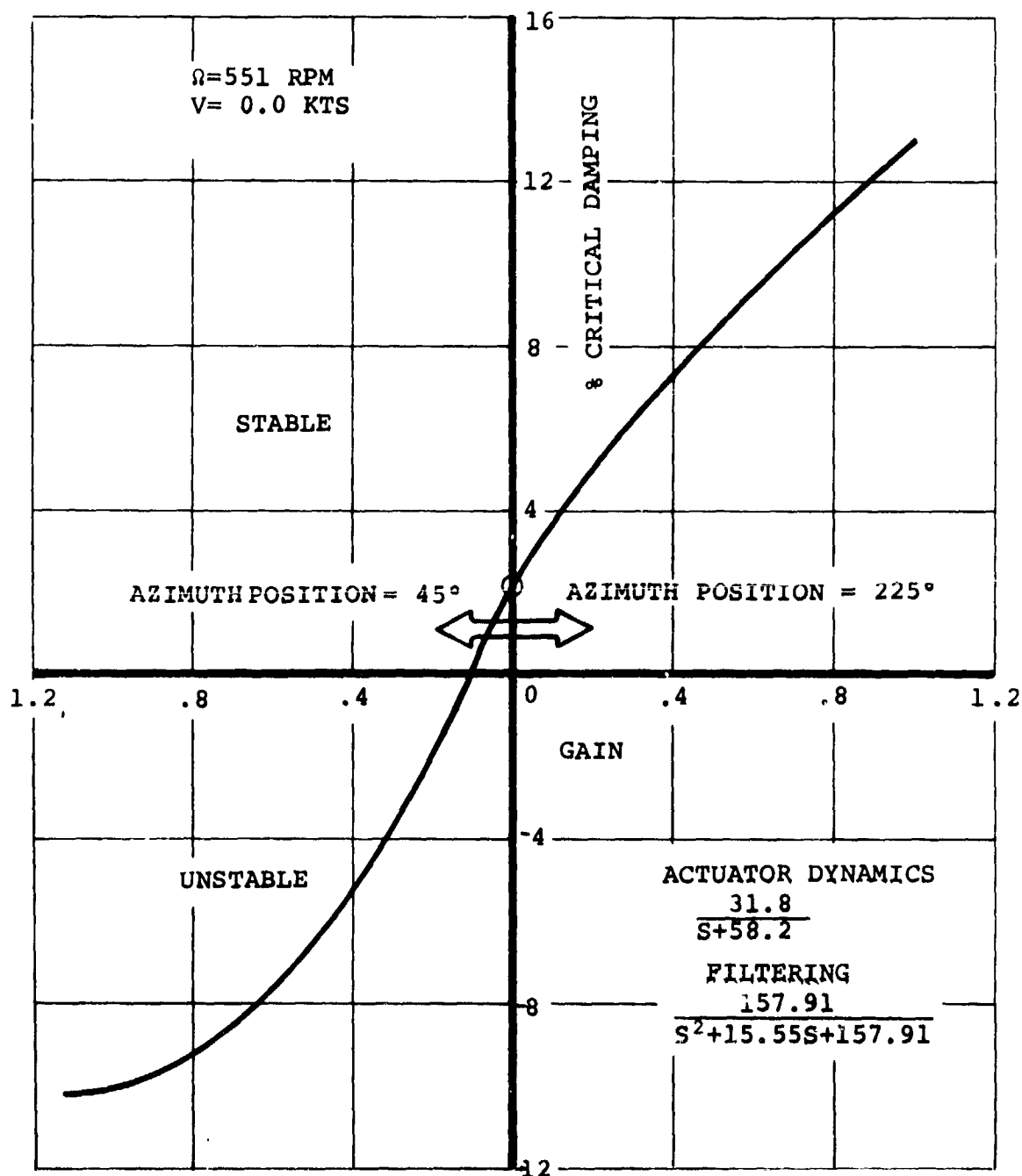
FIGURE 6.2.   FEEDBACK EFFECT ON THE LOWER
LEAD-LAG MODE
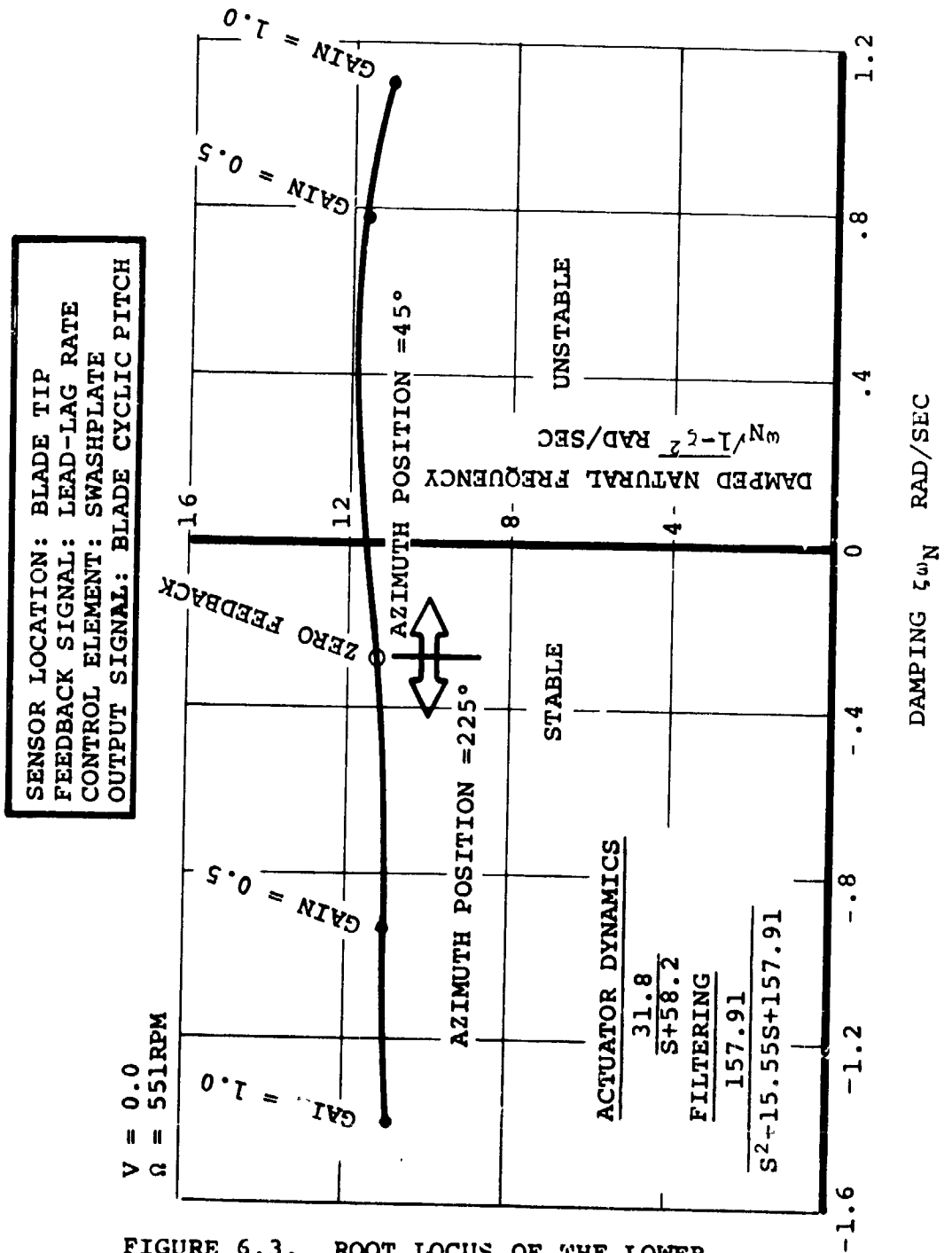
FIGURE 6.3.   ROOT LOCUS OF THE LOWER
LEAD-LAG MODE

SENSOR LOCATION: BLADE TIP
FEEDBACK SIGNAL: BLADE LEAD-LAG RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH

Ω=551 RPM
V=0.0
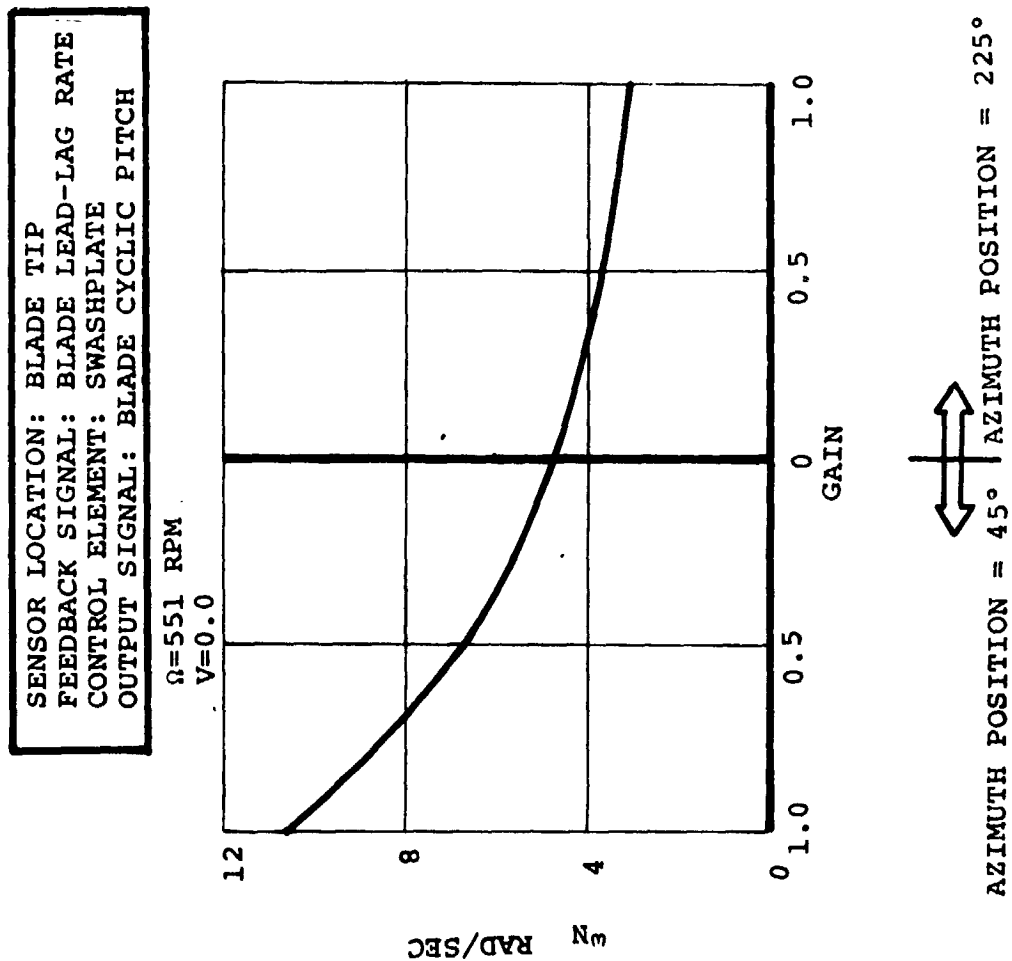
FIGURE 6.4.   FEEDBACK EFFECT ON PITCH ATTITUDE

follows an arc with the radius centered at the origin). In
fact, there is a phase shift which, if removed, would enable
greater values of critical damping to be obtained at lesser
values of gain. Degradation of the aircraft characteristics
is noted in the root associated with the aircraft pitch
degree of freedom. As the gain is increased to the maximum
investigated an approximate 30% reduction in the pitch
attitude damping coefficient was observed. This is an un-
desirable by-product which can be eliminated by the proper
design of a band-pass filter.

Although the study only entered into limited detail, a
significant increase in damping was observed, and it is
clear that refinement of azimuth position, phasing and
filtering would enable further increases in damping to be
obtained.

6.4 EVALUATION OF SYSTEM SENSING HUB IN-PLANE VELOCITY

The study then proceeded to a system using hub in-plane
velocity as the feedback signal. Representative actuator
and filter transfer functions were used to give some feel
for the practical aspects of the problem. Hub in-plane
velocity was selected because the $(\Omega - \omega_L)$ mode of the blade
will produce deflections in the plane of the rotor and there
is, therefore, some expectation that this signal will produce
a similar result when fed back. An accelerometer was mounted

on the hub in a manner to sense hub in-plane acceleration.
This signal was integrated to obtain hub in-plane velocity and
filtered through a low-pass filter (break frequency at 2 cps)
to attenuate the high frequency signals. Four azimuth
positions were evaluated: 45-degrees, 124-degrees, 225-degrees,
and 304-degrees.

From the evaluation of each of these azimuth positions it was
hoped to define the proper quadrant for azimuth position and
to obtain a feeling for the optimum gain range. Figures 6.5
through 6.10 delineate the results of the study. For all
azimuth positions and gains the system appeared to be very
ineffective in augmenting the damping of the $(\Omega-\omega_L)$ mode.

There was a strong effect on the pitch attitude and a coupled
vertical bending mode which can be eliminated by proper
filtering. The investigation was suspended at this point.
It is believed that with further refinement the design of
the feedback loops (with strict attention to the phase re-
lationships) a significant increase in the damping could be
obtained.

6.5 CONCLUSIONS

Damping in the lower lead-lag mode $(\Omega-\omega_L)$ may be increased by
the use of feedback as shown by the results for the ideal
system studied. Design of system for operational use will
entail further study of sensors, signals and feedback loops.
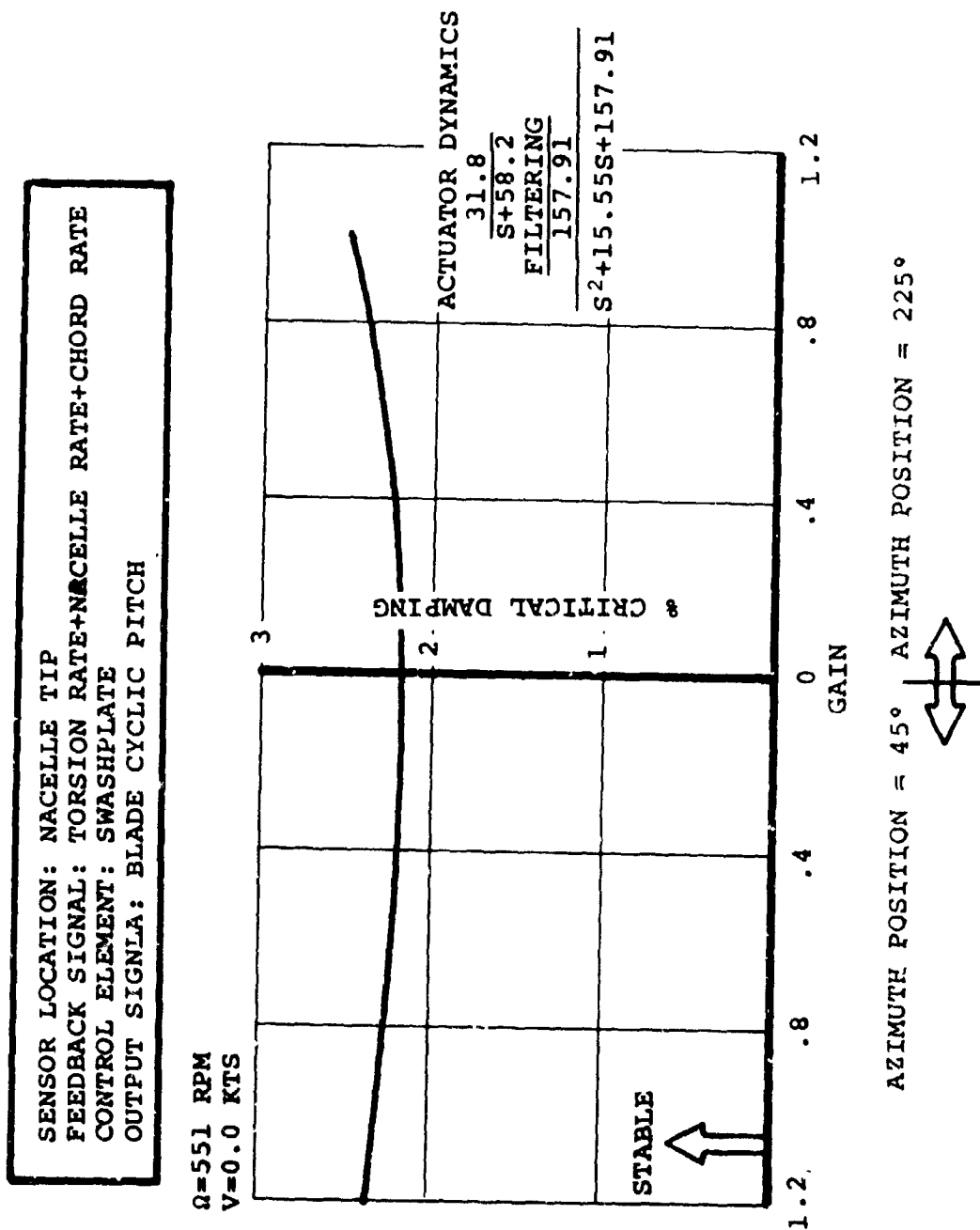
FIGURE 6.5.   FEEDBACK EFFECT ON LOWER
LEAD-LAG MODE

SENSOR LOCATION: NACELLE TIP
FEED BACK SIGNAL: TORSION RATE + CHORD RATE
                                   + NACELLE RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH

Ω = 551 RPM
V = 0.0

AZIMUTH POSITION = 45° | AZIMUTH POSITION = 225°
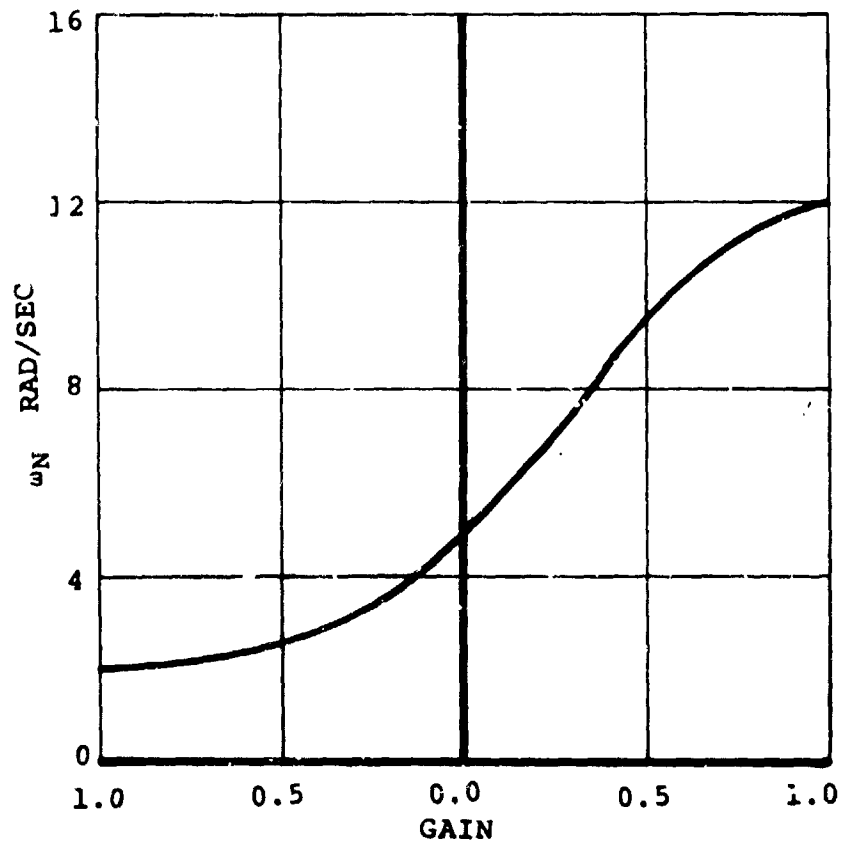
FIGURE 6.6 FEEDBACK EFFECT ON PITCH ATTITUDE

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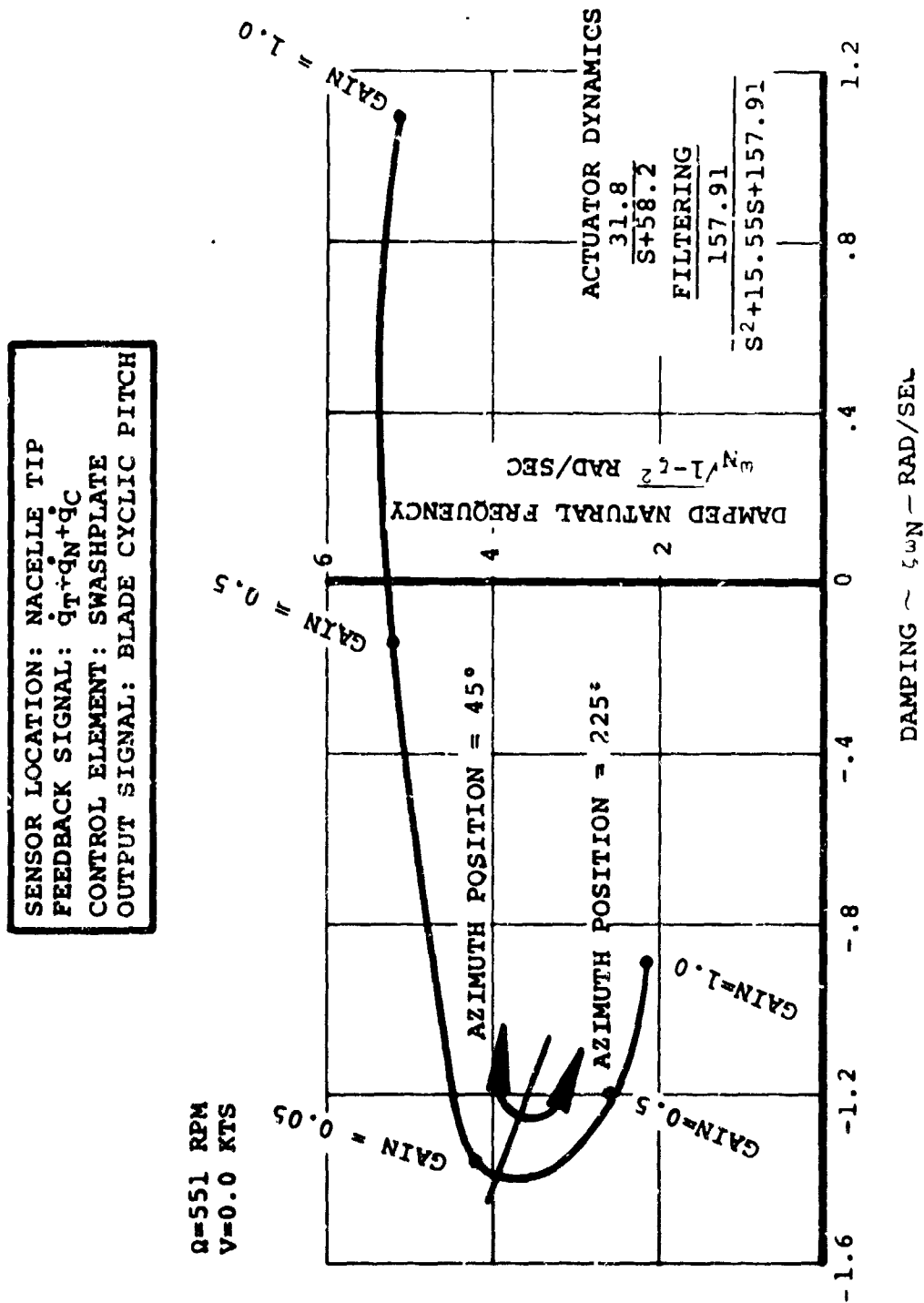
FIGURE 6.7.   ROOT LOCUS OF COUPLED VERTICAL
BENDING/LEAD-LAG MODE

SENSOR LOCATION: NACELLE TIP
FEEDBACK SIGNAL: TORSION RATE + NACELLE RATE + CHORD RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH

Ω=551 RPM
V=0.0 KTS

ACUTATOR DYNAMICS
$\frac{31.8}{S + 58.2}$

FILTERING
$\frac{157.91}{S^2 + 15.55S + 157.91}$

% CRITICAL DAMPING

GAIN

STABLE

AZIMUTH POSITION = 124°   AZIMUTH POSITION = 304°
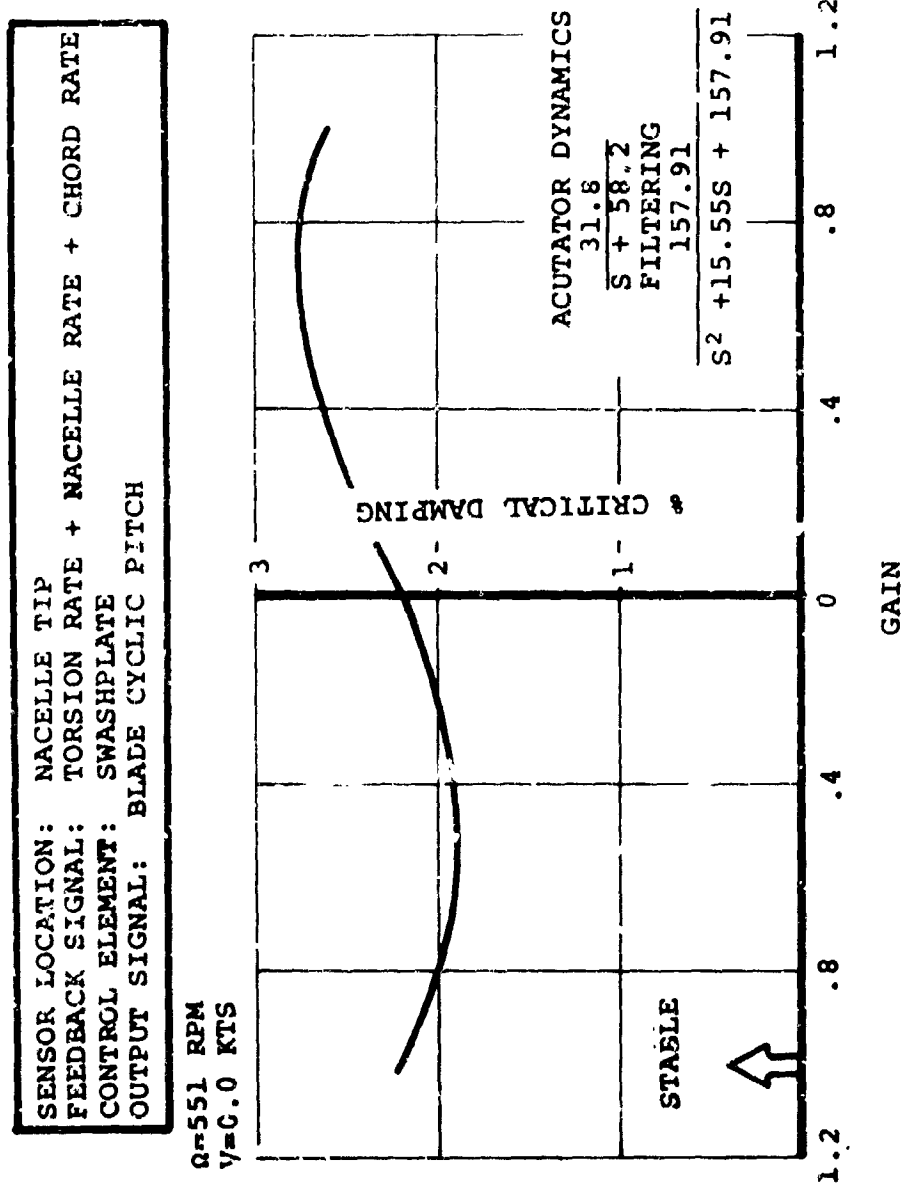
FIGURE 6.8.   FEEDBACK EFFECT ON THE L. .R
LEAD-LAG MODE

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SENSOR LOCATION: NACELLE TIP
FEEDBACK SIGNAL: TORSION RATE + CHORD RATE
+ NACELLE RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH

Ω=551 RPM
V=0.0
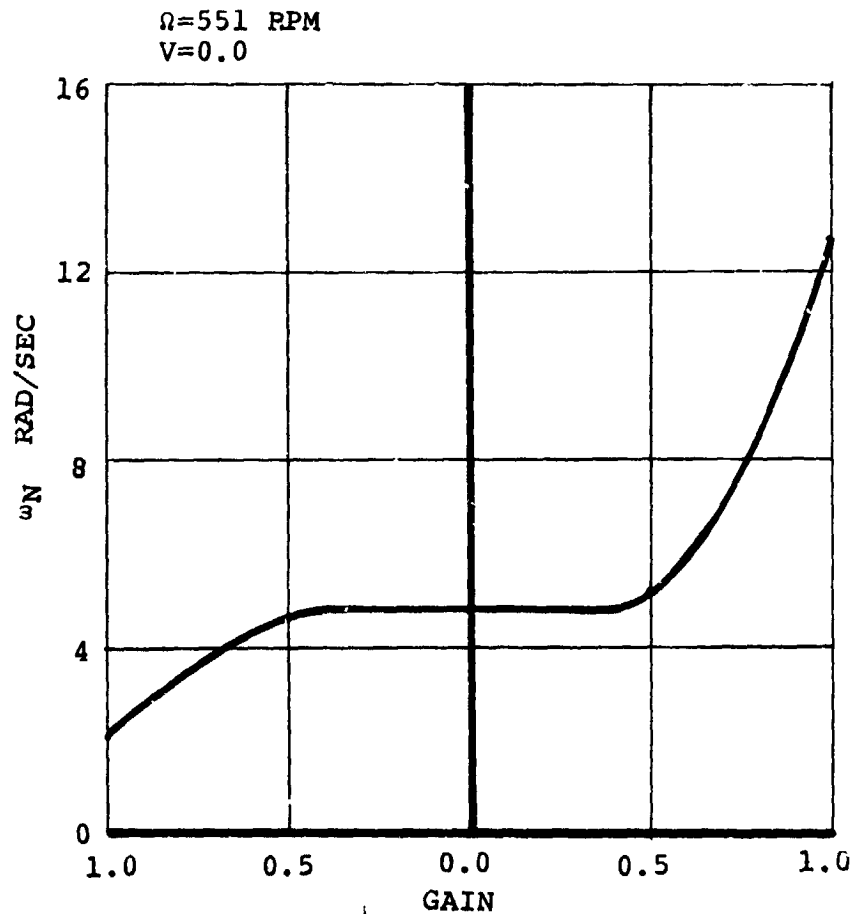
AZIMUTH POSITION = 124°   AZIMUTH POSITION = 304°

FIGURE 6.9. FEEDBACK EFFECT ON PITCH ATTITUDE

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SENSOR LOCATION: NACELLE TIP
FEEDBACK SIGNAL: TORSION RATE + NACELLE PITCH RATE
+ CHORD RATE

CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: BLADE CYCLIC PITCH

Ω=551 RPM
V=0.0 KTS

ACTUATOR DYNAMICS
$\dfrac{31.8}{S + 58.2}$
FILTERING
$\dfrac{157.91}{S^2 + 15.55S + 157.91}$

DAMPING NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$

GAIN = 1.0

ZERO FEEDBACK

GAIN=+.5

GAIN=1.0

DAMPING ~ $\zeta\omega_N$ ~ RAD/SEC

AZIMUTH POSITION = 304°   AZIMUTH POSITION = 124°
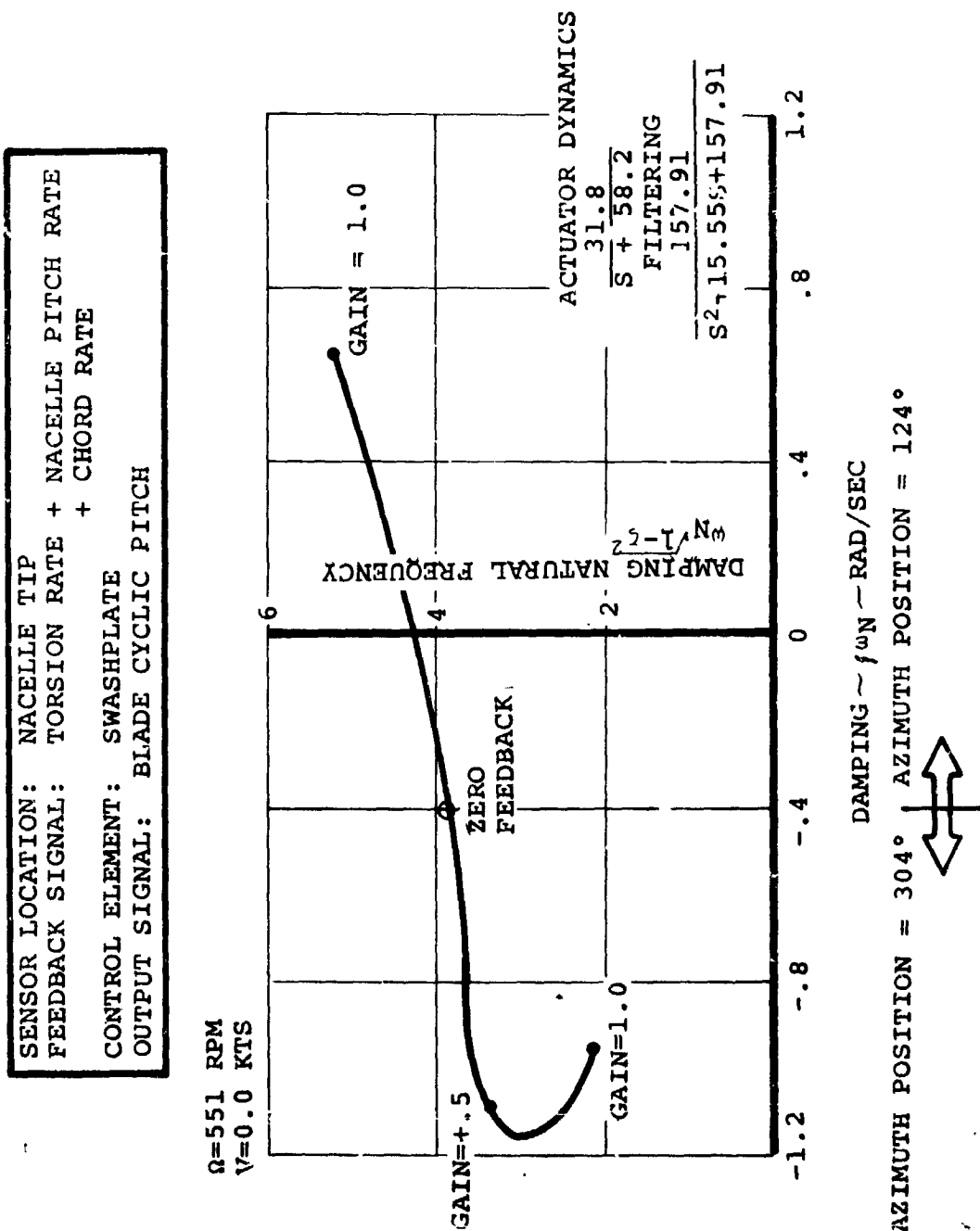
FIGURE 6.10.   ROOT LOCUS OF COUPLED VERTICAL
BENDING/LEAD-LAG MODE

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7.   THRUST MANAGEMENT AND ROTOR GOVERNING FEEDBACK SYSTEMS

7.1   BACKGROUND

Although the study of governing systems was not specifically
a part of the work statement of this contract, the study of
feedback systems to minimize aircraft response to axial gusts
was included in the statement of work. Since the rotor and
aircraft response to axial gusts can be completely changed by
changes in governing systems, and in fact the governor can
itself be the feedback system to minimize gust response, it
was necessary to study the governing system as a whole.

This section therefore addresses the problems associated with
the design of a thrust management and rotor governing system
for a tilt rotor aircraft with large diameter flexible rotors
which are lightly loaded in cruise. The power plants are
located close to the rotors and cross shafting is provided so
that a single power failure can be accommodated.

In a helicopter, rapid response to pilot demand for changes in
thrust is of paramount importance to permit accurate height
control in hover and near vertical flight. This requirement
has resulted in helicopters with direct pilot control of collec-
tive pitch, with the governor maintaining rpm by governing
fuel flow. In a turboprop airplane, the pilot wants to be
able to demand a fixed power, (e.g. max power throughout take-
off, or normal rated power throughout climb) and so turboprop

airplanes give the pilot direct control of power (fuel flow)
and the governor maintains rpm by governing blade pitch.

A V/STOL aircraft such as a tilt rotor which flies in both
helicopter and fixed wing modes needs to meet both of these
requirements, so that either both types of governing systems
must be installed with some changeover from one to the other
as flight mode is varied, or one system must be made to meet
satisfactorily the requirement of all regimes of flight.

Use of pilot demand of collective in cruise is considered un-
acceptable because the extreme sensitivity of power to collec-
tive pitch in this regime appears to make it impractical to
give the pilot adequate control sensitivity. Installation of
two separate systems is undesirable from the point of view
of complexity and pilot work load.

Adaptation of the fixed wing airplane system of power demand
and collective governing to the helicopter mode can be achieved
by making a mechanical demand for collective pitch coincident
with the power demand and having the governor perform only a
trimming function on the mechanically demanded collective.
Such a system is used on the Canadair CL-84, and has received
favorable pilot comment. This system has therefore been selec-
ted for tilt rotor.

7.2 STUDY APPROACH

The study uses three methods of evaluation:

O   inner loop response

O   outer loop response

O   gust response

Inner loop response concerns the RPM response when separated from the aircraft rigid body modes; that is, the aircraft is restrained from moving longitudinally, laterally, vertically and in pitch, roll or yaw. Pertinent structural modes and blade elastic modes remain in the model (See Appendix A for description of math model). Use of this method simplifies the studies and allows the analyst to evaluate the governor only. In the outer loop response all inner loop degrees of freedom are maintained and the aircraft rigid body modes of interest are added to the model. With this technique the analyst knows the governor characteristics from the inner loop response and can concentrate on evaluating the effect on aircraft flying qualities (such as dutch roll response in cruise or vertical response in hover). Gust response evaluation uses the outer loop model and "hits" it with a gust of known magnitude and shape. Time history responses are obtained and effect of the governor on aircraft ride qualities and structural response may be evaluated.

Using these techniques of evaluation three governor configurations were studied:

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O   single governor with one sensor (Figure 7.1)

O   dual independent governors (Figure 7.2)

O   single governor with sensors on each rotor (Figure 7.3)

Two modes of operation were considered for each governor. The
all operational mode (i.e., all aircraft components operational
and performing as designed) and the cross-shaft failure mode
(i.e., the cross-shaft connecting the two rotors has been
severed and they are free to rotate independently).

The study has been divided into four sections. The first
deals with the criteria developed for evaluation of the
governor on a tilt rotor aircraft. The second section evaluates
the three governor configurations and a final configuration is
chosen. Third, the effect of the governor on aircraft flying
qualities is evaluated and lastly the aircraft gust response
for conditions of no governor, fuel governor and collective
pitch governor are analyzed.

7.3 DESIGN CRITERIA

Governing accuracy criteria were set up, based on fixed wing
propeller airplane experience. The criteria used were:

O Maximum error of .3% within 3 seconds
   of a disturbance

O Maximum overshoot of 2% rpm in the transient
   response to a step input

Fulfillment of these goals requires that the RPM variation

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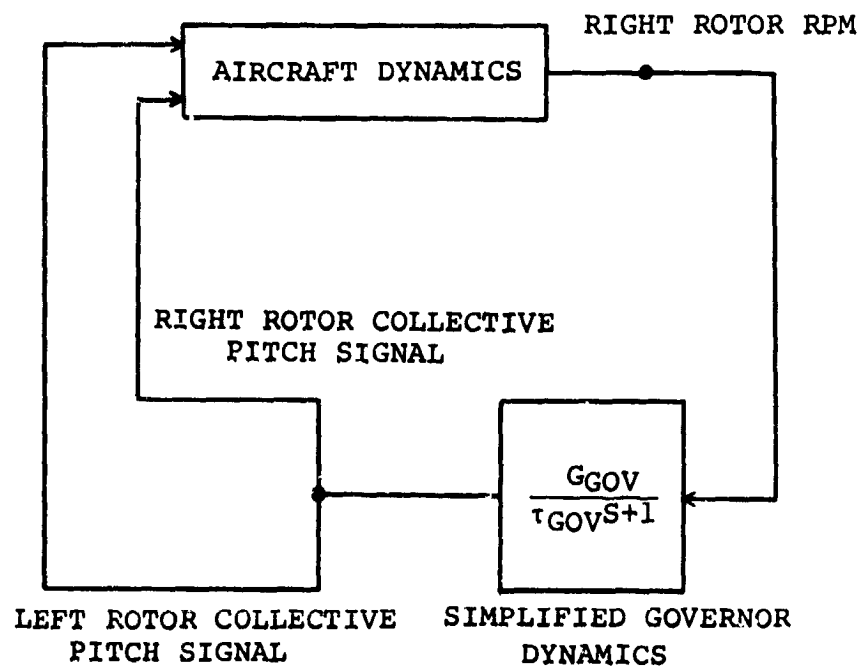
FIGURE   7.1. CANDIDATE GOVERNOR SCHEMATIC
SINGLE GOVERNOR WITH ONE SENSOR
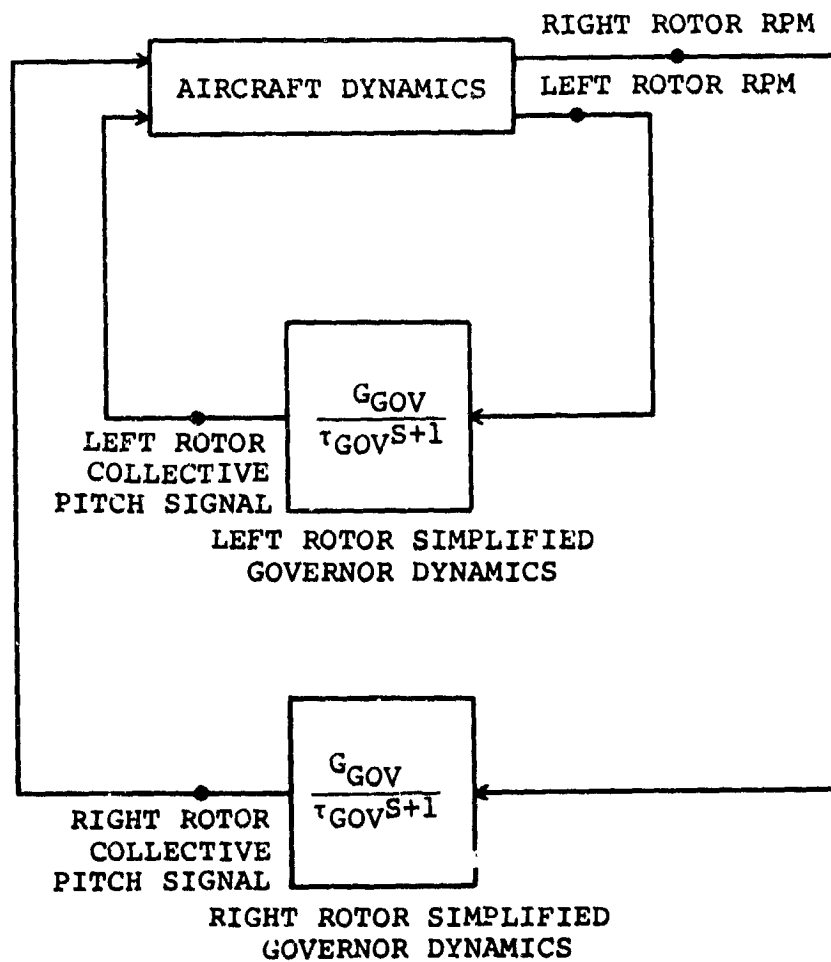
FIGURE 7.2. CANDIDATE GOVERNOR SCHEMATIC
DUAL INDEPENDENT GOVERNORS
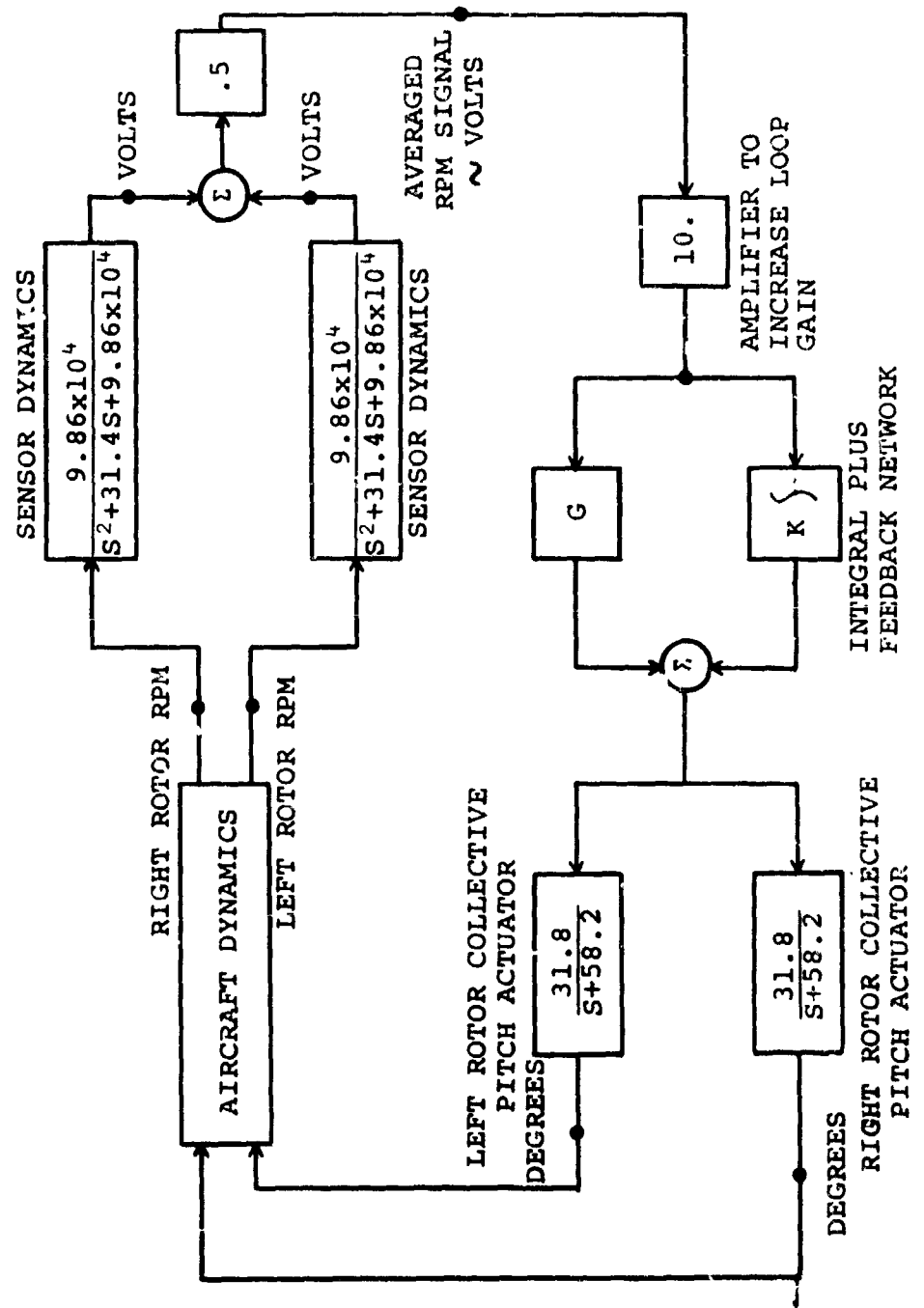
FIGURE 7.3. CANDIDATE GOVERNOR SCHEMATIC
SINGLE GOVERNOR WITH TWO SENSORS

mode has a frequency of 3.0 radians per second and a damping
coefficient of .8. These requirements should be fulfilled in
the all-operational mode and the cross-shaft failure mode.

Flying qualities of the aircraft should not be adversely
affected; that is, the governor configuration should not affect
the rigid body modal response in such a manner that they do
not meet the applicable specification requirements. The flying
qualities portion of the study will concentrate on evaluating
the effect of the governor on:

O vertical damping in hover

O roll damping in hover

O dutch roll damping in cruise

O phugoid damping in cruise

It is the goal of this study to reduce the axial gust response
of the rotor to a near zero level, however, if this is not
achievable axial accelerations of .2"g" will be tolerated for
a gust velocity of 20 fps. This level of gust response is a
result of studies performed for the SST to determine objection-
able threshold gust response characteristics in cruise.

7.4 EVALUATION OF COLLECTIVE PITCH GOVERNOR CONFIGURATIONS

Evaluation of the three governor configurations was performed
to determine which would best suit the needs of a tilt rotor
aircraft. Inner loop studies were made for all configurations

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and some limited outer loop studies were made when necessary. The all-operational and cross-shaft failure modes were considered for each configuration.

7.4.1 Single Governor With One Sensor

As shown in Figure 7.1, the single governor with one sensor would sense rotor RPM of either the left or right rotor (in this case the right rotor) and use the error signal to drive collective pitch for governing of both rotors. In the all-operational mode this governing concept is acceptable in that cross-shafting forces the rotors to rotate in unison and rotor RPM is essentially identical. However, if the cross-shaft fails the rotors are spinning independently; therefore, only the side with the sensor is being governed. This effect can be seen from the stability root variation with gain of Figure 7.4. The root associated with the left rotor RPM is not affected by changing governor gain which indicates that this rotor is not being governed. The operation depicted by the root is unacceptable and disqualifies this configuration from further consideration.

7.4.2 Dual Independent Governors

The dual independent governor configuration (Figure 7.2) operates on each rotor separately with no inter-rotor

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SINGLE GOVERNOR - 1 SENSOR (RT. ROTOR)
HOVER INNER LOOP
RPM FEEDBACK
$\tau = 0.0$
NO CROSS SHAFT

GAIN ~ RAD/RAD/SEC

-.005   0   .005   .01   .015

NOTE: THIS FEEDBACK HAS
NO EFFECT ON THE LEFT ROTOR
STABILITY ROOTS

RIGHT ROTOR
STABILITY ROOT

LEFT ROOT STABILITY ROOT

1.5

1.0

.5

DAMPED NATURAL FREQUENCY, $w_N \sqrt{1-\zeta^2}$ (RAD/SEC)

-1.5   -1.0   -.5   0   .5

DAMPING COEFFICIENT, $\zeta\omega$ (RAD/SEC)
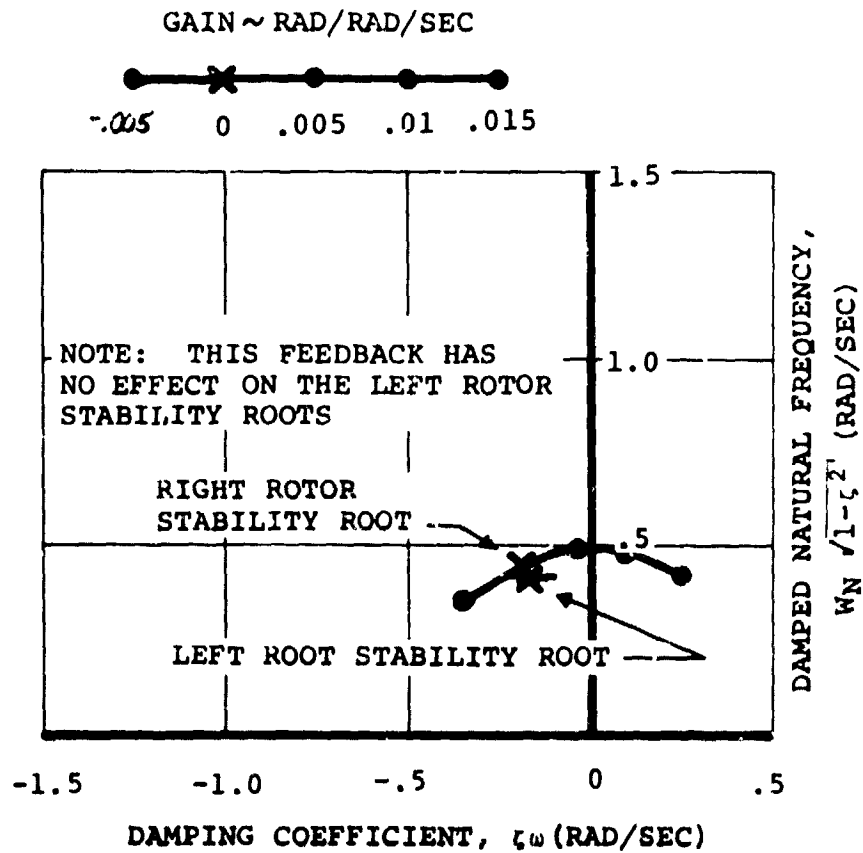
FIGURE 7.4. SINGLE GOVERNOR WITH ONE SENSOR -
STABILITY ROOT VARIATION

coupling. Figures 7.5 through 7.9 illustrate that with the cross-shaft out both rotors are governed and the problem seen with the single governor with one sensor does not appear. Note the triangles in Figure 7.5 showing that the presence of the cross-shaft has no effect on the rotor rotation mode root migration. The effects of RPM, integral of RPM and RPM rate feedback were explored for various governor time constants and no problems arose with other modes going unstable.

Although the gains chosen to evaluate this configuration were not of sufficient level to meet the error criteria, the trends are important to note. RPM feedback increases the damping of the rotor rotation mode while keeping the natural frequency constant (for a governor time constant of 0.0). Figure 7.7 illustrates this affect. Notice that the root migration follows an arc of approximately constant radius (constant natural frequency). As the roots follow the arc toward the negative real axis the damping ratio is increased. Also from this figure the destabilizing effect of governor time constant is seen. As time constant is increased the locus of roots is rotated in such a manner that the increase in damping ratio per unit gain is decreased. Integral of RPM feedback (Figure 7.8) increases the damped natural frequency while maintaining

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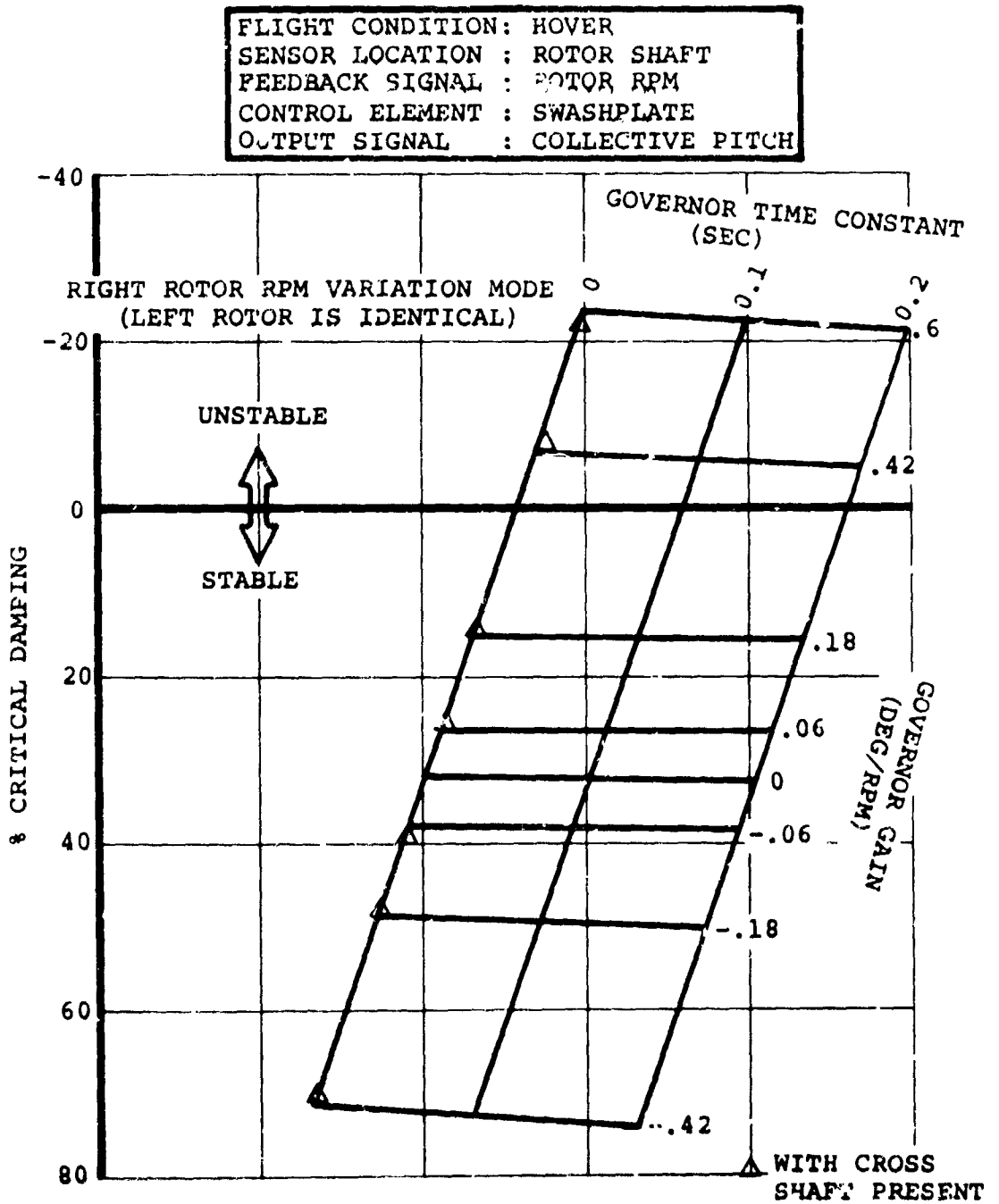
FIGURE 7.5.   INNER LOOP STABILITY VARIATION FOR
DUAL INDEPENDENT GOVERNORS WITHOUT
CROSS SHAFT

FLIGHT CONDITION: HOVER
SENSOR LOCATION : ROTOR SHAFT
FEEDBACK SIGNAL : INTEGRAL OF RPM
CONTROL ELEMENT : SWASHPLATE
OUTPUT SIGNAL : COLLECTIVE PITCH

RIGHT ROTOR
RPM VARIATION MODE
(LEFT ROTOR IS IDENTICAL)
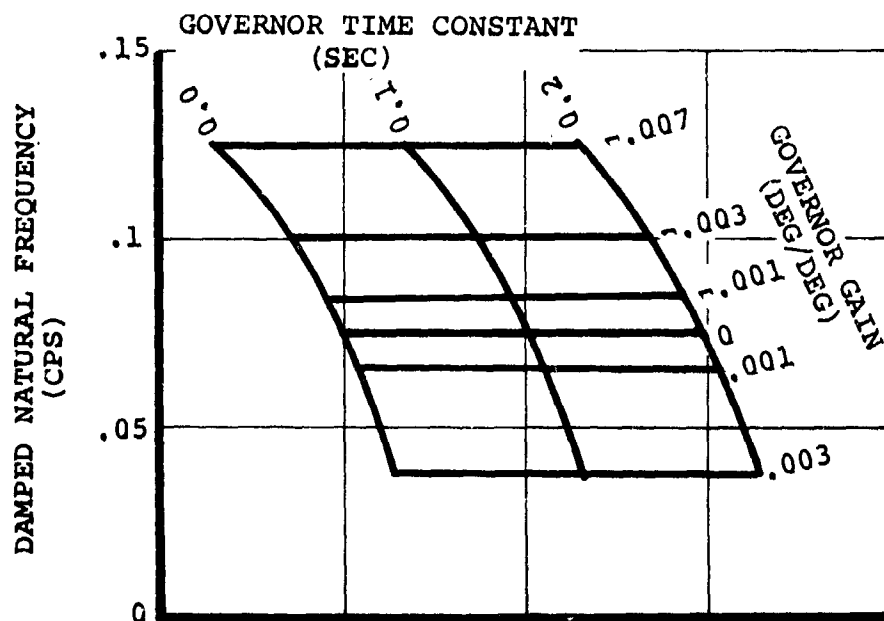
FIGURE 7.6. INNER LOOP STABILITY VARIATION FOR
DUAL INDEPENDENT GOVERNORS WITHOUT
CROSS SHAFT

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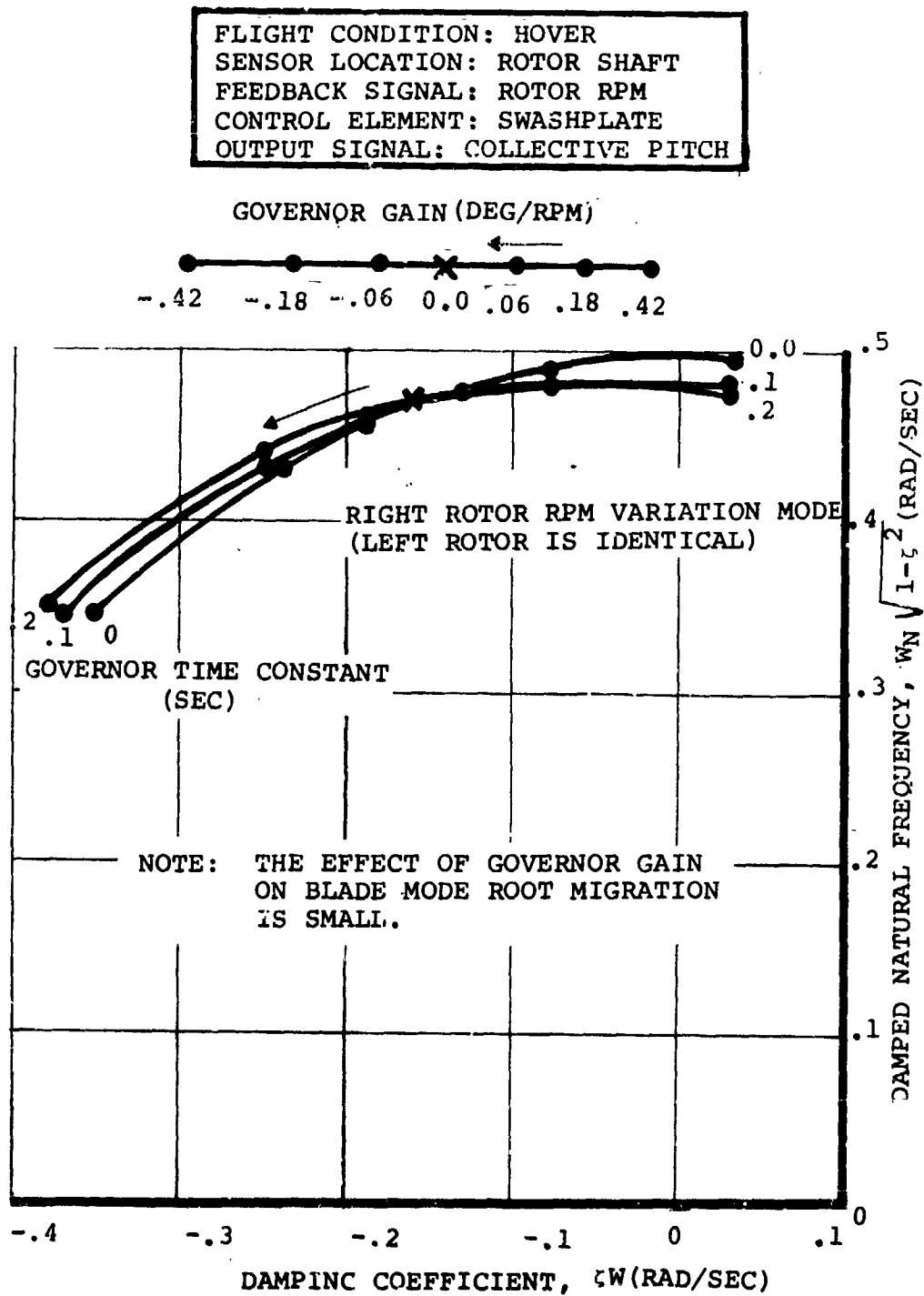
FIGURE 7.7. INNER LOOP STABILITY VARIATION FOR
DUAL INDEPENDENT GOVERNORS WITHOUT
CROSS SHAFT

FLIGHT CONDITION: HOVER
SENSOR LOCATION: ROTOR SHAFT
FEEDBACK SIGNAL: INTEGRAL OF ROTOR RPM
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: COLLECTIVE PITCH

GOVERNOR GAIN (DEG/DEG)

-.007   -.003   -.001   0   .001   .003

RIGHT ROTOR
RPM VARIATION MODE
(LEFT ROTOR IS IDENTICAL)

NOTE: THE EFFECT OF
GOVERNOR GAIN ON
BLADE MODE ROOT
MIGRATION IS SMALL

.2 .1 .0

GOVERNOR TIME
CONSTANT (SEC)

DAMPED NATURAL FREQUENCY $w_N \sqrt{1-\zeta^2}$ (RAD/SEC)

.8

.6

.4

.2

0

-.8   -.6   -.4   -.2   0   .2

DAMPING COEFFICIENT, ζW (RAD/SEC)

FIGURE 7.8. INNER LOOP STABILITY VARIATION FOR DUAL
INDEPENDENT GOVERNORS WITHOUT CROSS SHAFT

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FLIGHT CONDITION: HOVER
SENSOR LOCATION: ROTOR SHAFT
FEEDBACK SIGNAL: ROTOR RPM RATE
CONTROL ELEMENT: SWASHPLATE
OUTPUT SIGNAL: COLLECTIVE PITCH

GOVERNOR GAIN (DEG/DEG/SEC^2)

-.007   -.003   0   .003   .007   .01

GOVERNOR TIME
CONSTANT (SEC)   .2
                    .1
                 0.0

RIGHT ROTOR RPM VARIATION MODE
(LEFT ROTOR IS IDENTICAL)

DAMPED NATURAL FREQUENCY, $W_N \sqrt{1-\zeta^2}$ (RAD/SEC)

-.6   -.4   -.2   0

DAMPING COEFFICIENT, ζW (RAD/SEC)
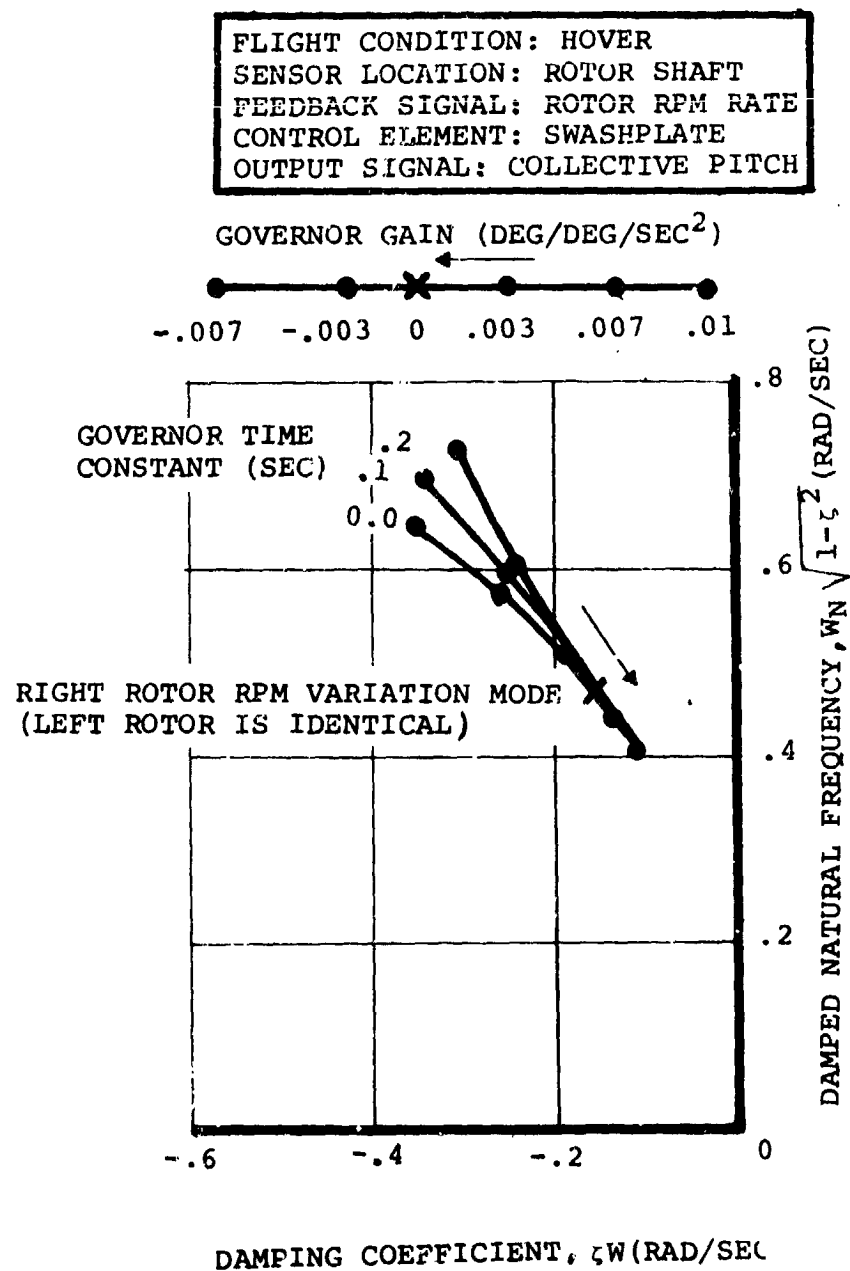
FIGURE 7.9. INNER LOOP STABILITY VARIATION FOR DUAL
INDEPENDENT GOVERNORS WITHOUT CROSS SHAFT

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a constant level of damping coefficient $(\zeta\omega_N)$. Once again the governor time constant is destabilizing in that the locus of roots is driven toward the unstable right half plane for negative feedback. RPM rate feedback (Figure 7.9) decreases the natural frequency and the damping coefficient. As time constant is increased the locus of roots is rotated to be similar to integral of RPM feedback. Analysis of these three graphs indicates that RPM and integral of RPM feedback will be useful in meeting the error criteria. RPM rate feedback is not as desirable for the task at hand.

Results of the inner loop study as discussed above indicated that the dual independent governor configuration could be acceptable to meet the steady state error and transient response criteria for the all-operational and cross-shaft failure mode. However, an earlier flying qualities study (NASA Design Study) indicated that the dual governor config- uration could have a deleterious effect on the roll mode in hover and the dutch roll mode in cruise. To confirm this trend the roll and vertical translation degrees of freedom were added to the hover math model and the studies of RPM and integral of RPM were repeated. Figures 7.10 and 7.11 show that adding these degrees of freedom have no effect on the inner loop response (i.e., the inner loop root migration is

DUAL GOVERNOR
HOVER - OUTER AND INNER LOOP STABILITY
RPM FEEDBACK
NO CROSS SHAFT
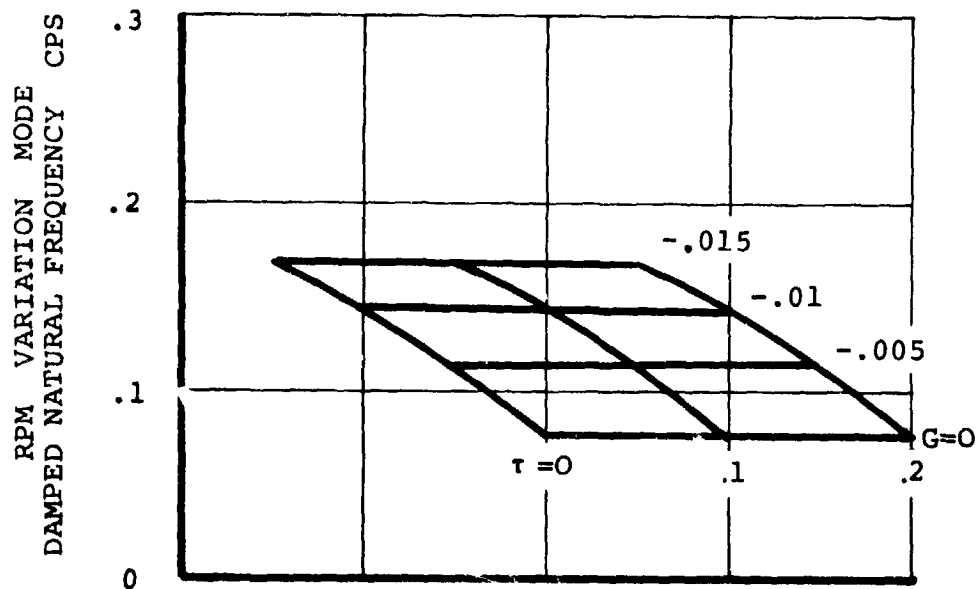
LEFT ROTOR

NOTES:
1. τ IS THE GOVERNOR TIME CONSTANT IN SECONDS
2. G IS THE GOVERNOR GAIN IN RAD/RAD/SEC

FIGURE 7.10. DUAL INDEPENDENT GOVERNORS - INNER AND OUTER LOOP STABILITY RESPONSE

DUAL GOVERNOR
HOVER - OUTER AND INNER LOOP STABILITY
RPM FEEDBACK
NO CROSS SHAFT
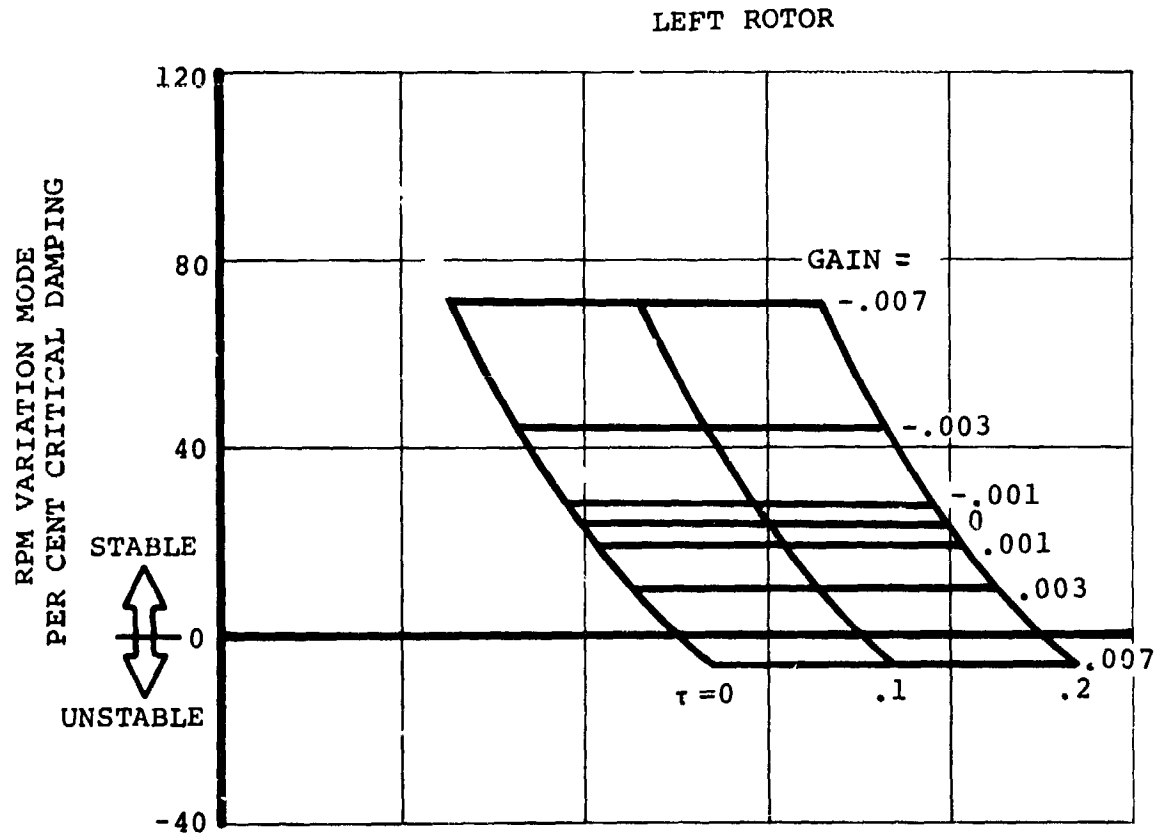
LEFT ROTOR

NOTES:

1. τ IS THE GOVERNOR TIME CONSTANT IN SECONDS
2. G IS THE GOVERNOR GAIN IN RAD/RAD/SEC

FIGURE 7.41. DUAL INDEPENDENT GOVERNORS - INNER AND OUTER LOOP STABILITY RESPONSE

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identical). Figures 7.12 and 7.13 show the rotor rotation

and rigid body root migration. Note that in each case both

the hover roll and hover vertical translation modes were

driven less stable (i.e., the real roots move from the left

half plane toward the origin). In cruise the same type of

effect would occur for the dutch roll mode and phugoid mode.

Although inner loop response for the dual independent governor

configuration is acceptable, the effect on the roll and dutch

roll modes is not and disqualifies this configuration.

7.4.3 Single Governor With Two Sensors

A schematic of the single governor with two sensors is shown

in Figure 7.3. In this configuration the RPM variation of

each rotor is sensed, the signals are averaged and collective

pitch is fed back as a function of the averaged signal. Since

this configuration was felt to be the most likely candidate

for an optimum tilt rotor governor system, the feedback loop

has been made considerably more complex. Sensor dynamics

have been added and are represented as a second order lag with

a break frequency of 50 cps. This is a conservative estimate

of sensor dynamics in that magnetic tachometers have a "near

instantaneous" response. Actuator dynamics are representative

of actuators used on Boeing Vertol test hardware. The

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DUAL GOVERNOR
HOVER - OUTER AND INNER LOOP STABILITY
RPM FEEDBACK
NO CROSS SHAFT

LEFT ROTOR

GAIN, RAD/RAD/SEC

-.01 -.007 -.003 0 .003 .00⁄ .01

NOTE:
1. FOR PROPER SIGN OF FEEDBACK ALL BLADE MODES ARE STABLE

RIGHT ROTOR ROTATION MODE

ROLL MODE

VERTICAL MODE

LEFT ROTOR ROTATION MODE

DAMPED NATURAL FREQUENCY, $W_N \sqrt{1-\zeta^2}$ (RAD/SEC)

-2.0   -1.5   -1.0   -.5   0   .5

DAMPING COEFFICIENT, $\zeta\omega$ (RAD/SEC)
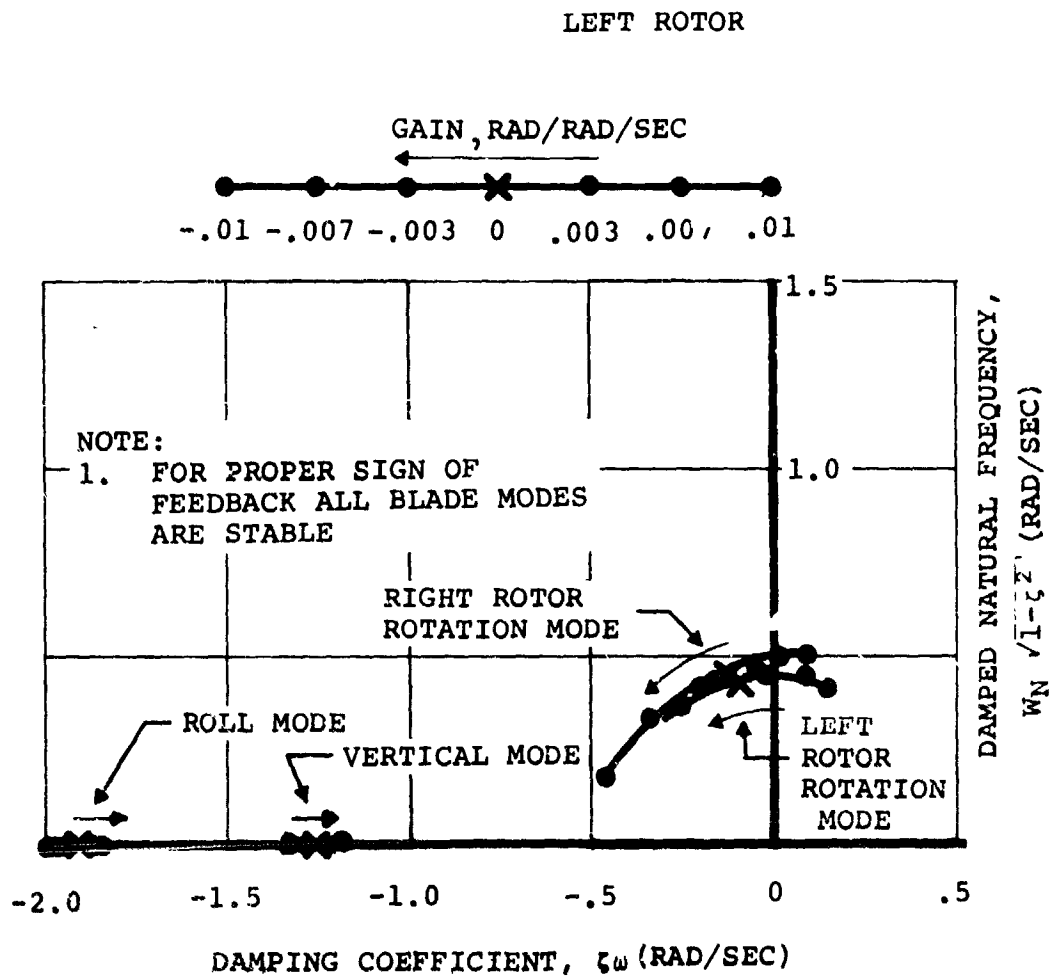
FIGURE 7.12. DUAL INDEPENDENT GOVERNORS - INNER AND OUTER LOOP STABILITY RESPONSE
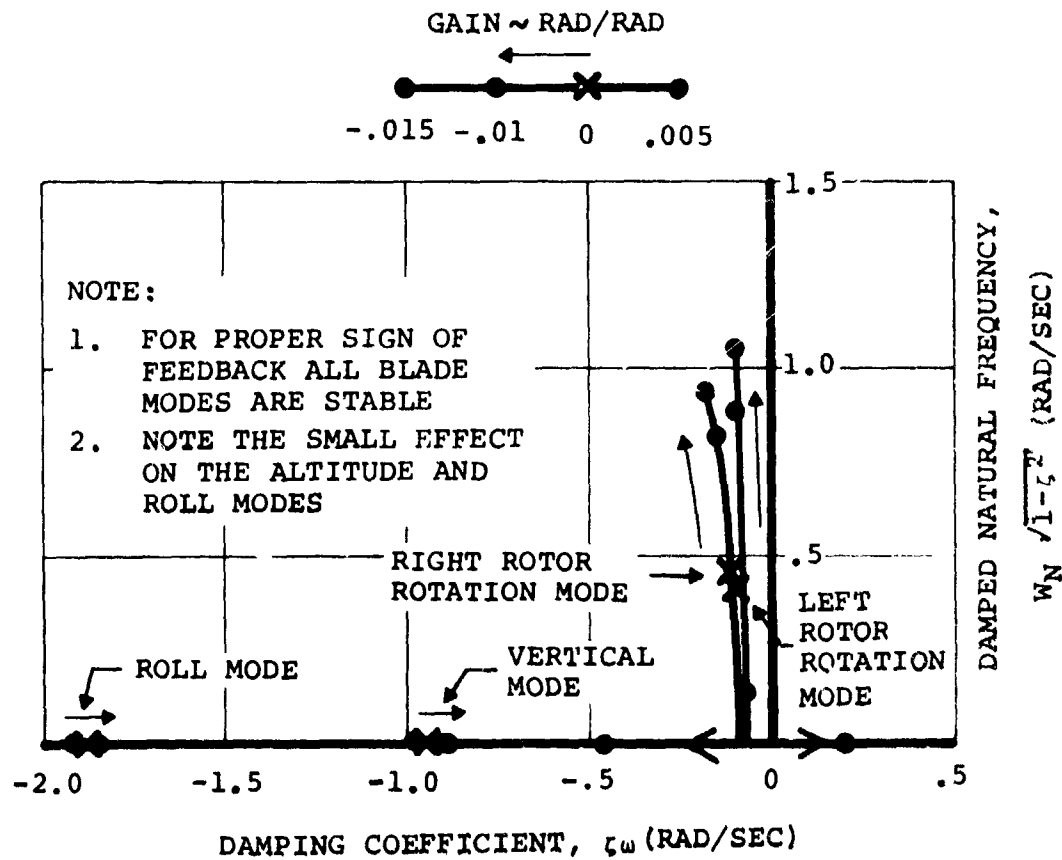
FIGURE 7.13. DUAL INDEPENDENT GOVERNORS - INNER
AND OUTER LOOP STABILITY RESPONSE

actuators are represented as first order lags with a time constant of .017 sec. Signal shaping has been added to the system as needed. With the addition of the above hardware to the feedback loop the governor representation is felt to be representative of an actual governor system.

Bode plots (Appendix B) were calculated for RPM feedback in various flight conditions in the all-operational mode and the cross-shaft failure mode to determine the maximum acceptable values of governor gain when designing to a 30° phase margin and a 6 dB gain margin. These plots are especially helpful as working tools in determining if there will be a stability problem. With them the need for filtering and the filter characteristics can be easily identified. The gains are shown in Figure 7.14 as a function of aircraft configuration, velocity and RPM. Maximum gain levels indicated that there would be no stability problem and that the necessary governor gain level would be well within the stability boundaries. For this reason bode plots were not calculated for integral of RPM feedback.

Figures 7.15 through 7.23 show the rotor rotation root migration with gain for:

    O   RPM and integral of RPM feedback

    O   cruise and hover modes
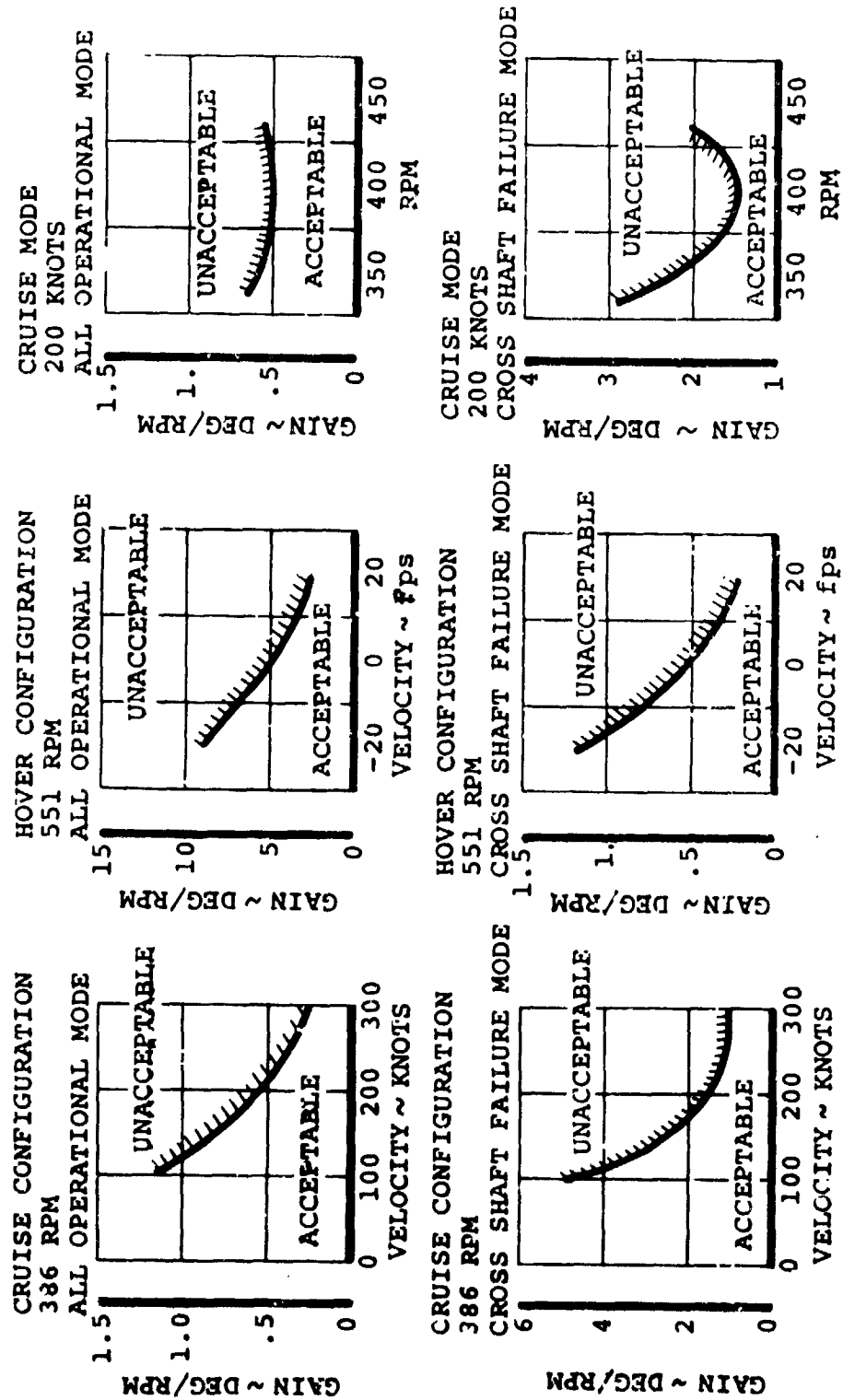
FIGURE 7.14. SINGLE GOVERNOR WITH TWO SENSORS
GOVERNOR GAIN LIMITS FOR RPM FEEDBACK
INNER LOOP STABILITY

o all-operational and cross-shaft failure modes

Note that in each case the governor has no effect on the anti-symmetric rotation mode. Although this is not the most desirable case from a purely RPM governing point of view, it is necessary to keep the hover roll damping and the cruise dutch roll damping from being degraded.

Initially, a straight proportional (RPM) feedback was evaluated (Figures 7.15, 7.16, 7.20 and 7.21). The desired damping ratio (shaping of transient response) could be achieved to meet the error criteria, but the natural frequency was too low in the hover regime (the desired accuracy could not be achieved) and in the cruise configuration the rotor rotation roots were real with one root very small, thus yielding poor transient response. To improve the accuracy in hover and the transient response in cruise, integral of RPM was fed back (Figures 7.17, 7.22 and 7.23). The gain was increased until the desired natural frequency was obtained. In the hover regime and the 100 knot cruise case, it was found necessary to have proportional as well as integral feedback (Figures 7.18 and 7.19) to achieve the desired accuracy and shaping of the transient response. The table below specifies the steady-state integral and proportional gains required to meet the accuracy and transient response goals.

227

HOVER CONFIGURATION
STEADY HOVER
551 RPM
CROSS SHAFT FAILURE MODE
GAINS

0.093.137.185

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$, RAD/SEC

1.0

.5

0

-1.0   -.5   0
DAMPING $(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps CLIMB
551 RPM
CROSS SHAFT FAILURE MODE
GAINS

0.065.109.164

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{-1\zeta^2}$, RAD/SEC

1.0

.5

0

-1.0   -.5   0
DAMPING $(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps DESCENT
551 RPM
CROSS SHAFT FAILURE MODE
GAINS

0.131.197.262

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$, RAD/SEC

1.0

.5

0

-1.0   -.5   0
DAMPING $(\zeta\omega_N)$, RAD/SEC

NOTES:

1. ● SYMMETRIC ROTATION MODE
   O ANTI-SYMMETRIC ROTATION MODE.
2. THIS GOVERNOR CONFIGURA-
   TION HAS NO EFFECT ON
   THE ANTI-SYMMETRIC
   ROTATION MODE.
3. NO MODES (EITHER ROTOR
   OR WING) ARE DRIVEN
   UNSTABLE WITH THE ABOVE
   GOVERNOR GAINS
4. SENSOR AND ACTUATOR
   DYNAMICS ARE INCLUDED
   IN THE MODEL.
5. WING VERTICAL BENDING
   AS WELL AS PERTINENT
   BLADE MODES ARE INCLUDED

FIGURE 7.15   SINGLE GOVERNOR WITH TWO SENSORS ROTOR ROTATION
               ROOT MIGRATION WITH   RPM FEEDBACK

228

FIGURE 7.16a. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION
WITH RPM FEEDBACK

HOVER CONFIGURATION
20 fps DESCENT
CROSS SHAFT PRESENT
551 RPM
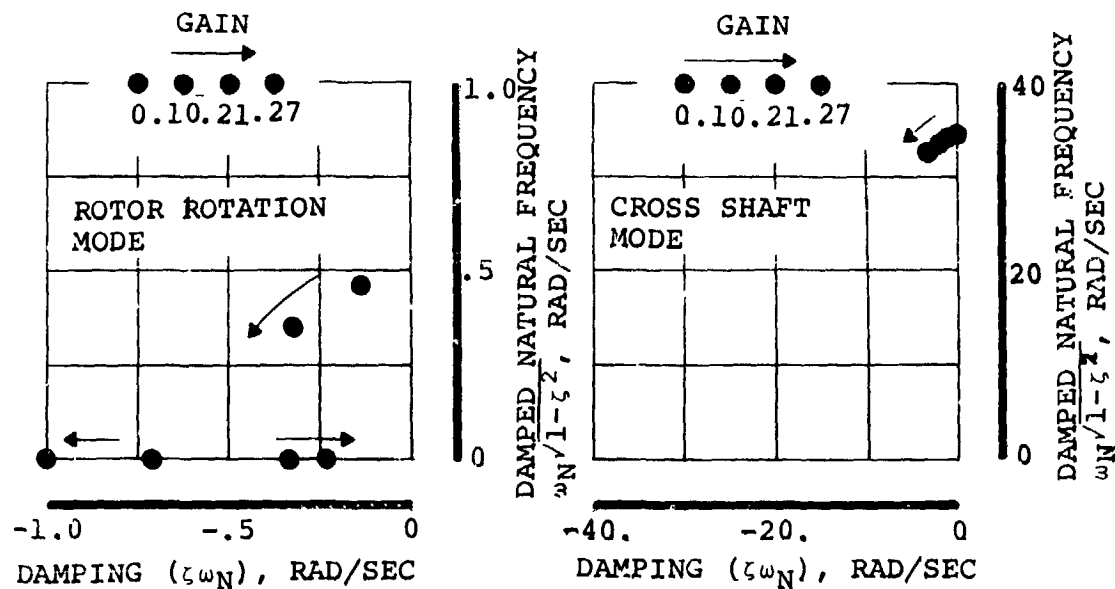
NOTES:

1. This gover... configuration has no effect on t' ross shaft (anti-symmetric ... rotation) mode.
2. No modes (ei...ar rotor or wing) are driven unstable with the above ...overnor gains.
3. Sen... and actuator dynamics are included in the model.
4. Wing vertical bending as well as pertinent blade mo...s are included in the mod...

FIGURE 7.16b.   ROTOR ROTATION ROOT MIGRATION
WITH RPM FEEDBACK

HOVER CCNFIGURATION
STEADY HOVER
551 RPM
CROSS SHAFT FAILURE MODE
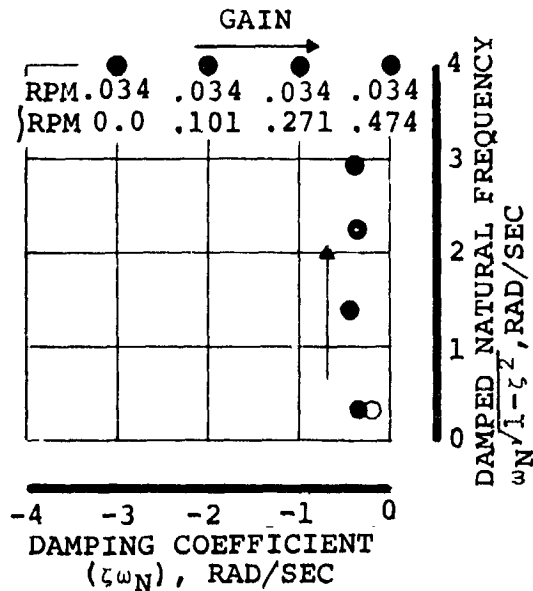
GAIN

RPM .034 .034 .034 .034
∫RPM 0.0 .101 .271 .474

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

DAMPING COEFFICIENT
$(\zeta \omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps CLIMB
551 RPM
CROSS SHAFT FAILURE MODE

GAIN

RPM .022 .022 .022 .022
∫RPM 0.0 .046 .175 .306

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

DAMPING COEFFICIENT
$(\zeta \omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps DESCENT
551 RPM
CROSS SHAFT FAILURE MODE

GAIN

RPM .049 .049 .049 .049
∫RPM 0.0 .147 .295 .393

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

DAMPING COEFFICIENT
$(\zeta \omega_N)$, RAD/SEC
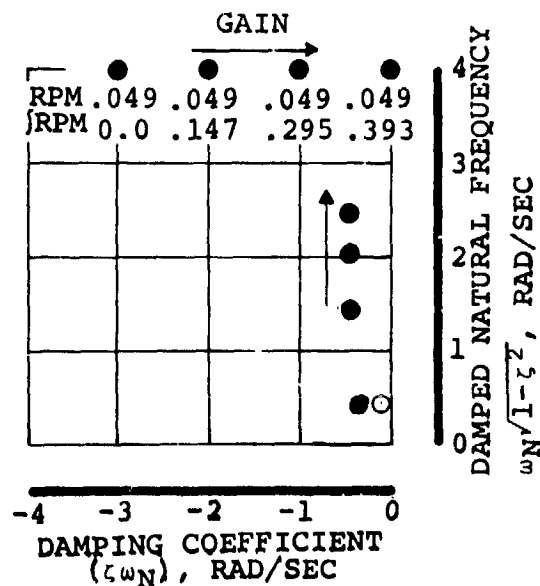
NOTES:
1. ● SYMMETRIC ROTATION MODE
   ○ ANTI-SYMMETRIC ROTATION MODE
2. THIS GOVERNOR CONFIGURATION HAS
   NO EFFECT ON THE ANTI-SYMMETRIC
   ROTATION MODE.
3. NO MODES ARE DRIVEN UNSTABLE
   WITH THE ABOVE GOVERNOR GAINS
4. SENSOR AND ACTUATOR DYNAMICS ARE
   INCLUDED IN THE MODEL.
5. WING VERTICAL BENDING AS WELL
   AS PERTINENT BLADE MODES ARE
   INCLUDED.

FIGURE 7.17. SINGLE GOVERNOR WITH TWO SENSORS
            ROTOR ROTATION RCOT MIGRATION WITH ∫+ RPM FEEDBACK

231

HOVER CONFIGURATION
STEADY HOVER
551 RPM
CROSS SHAFT FAILURE MODE

GAINS →

| RPM | .032 | .065 | .169 | .267 |
| ∫RPM | .469 | .469 | .469 | .469 |

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$, RAD/SEC

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps CLIMB
551 RPM
CROSS SHAFT FAILURE MODE

GAINS →

| RPM | .021 | .043 | .109 | .174 |
| ∫RPM | .305 | .305 | .305 | .305 |

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$, RAD/SEC

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps DESCENT
551 RPM
CROSS SHAFT FAILURE MODE

GAINS →

| RPM | .049 | .147 | .196 | .245 |
| ∫RPM | .393 | .393 | .393 | .393 |

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

NOTES:
1. ● SYMMETRIC ROTATION MODE
   ○ ANTI-SYMMETRIC MODE
2. THIS GOVERNOR CONFIGURATION
   HAS NO EFFECT ON THE ANTI-
   SYMMETRIC ROTATION MODE
3. NO MODES ARE DRIVEN UNSTABLE
   WITH THE ABOVE GOVERNOR GAINS
4. SENSOR AND ACTUATOR DYNAMICS
   ARE INCLUDED IN THE MODEL
5. WING VERTICAL BENDING AS WELL
   AS PERTINENT BLADE MODES ARE
   INCLUDED

FIGURE 7.18. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION WITH ∫ + RPM FEEDBACK
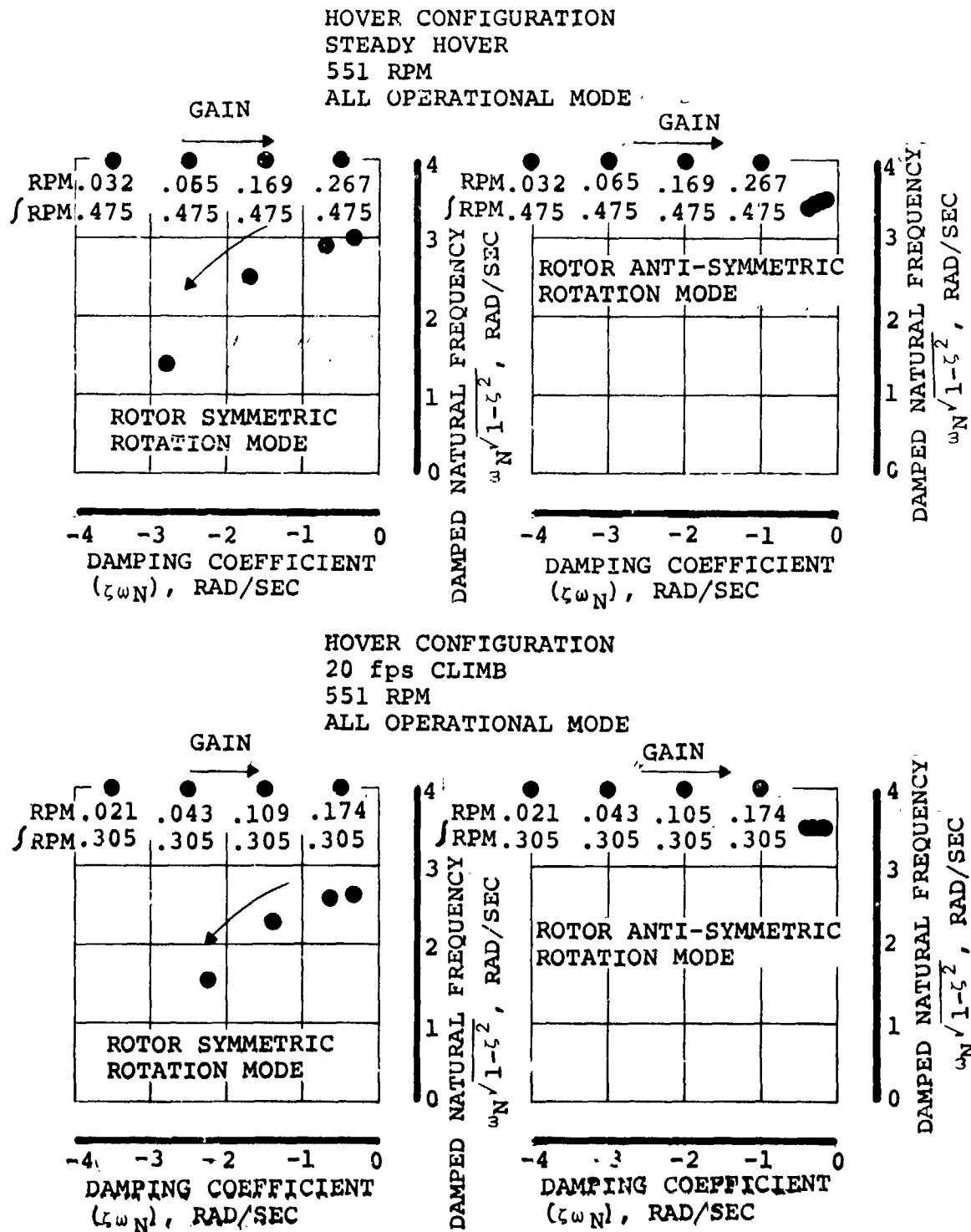
FIGURE 7.19a. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION WITH∫+RPM FEEDBACK

233

HOVER CONFIGURATION
20 fps DESCENT
551 RPM
ALL OPERATIONAL MODE
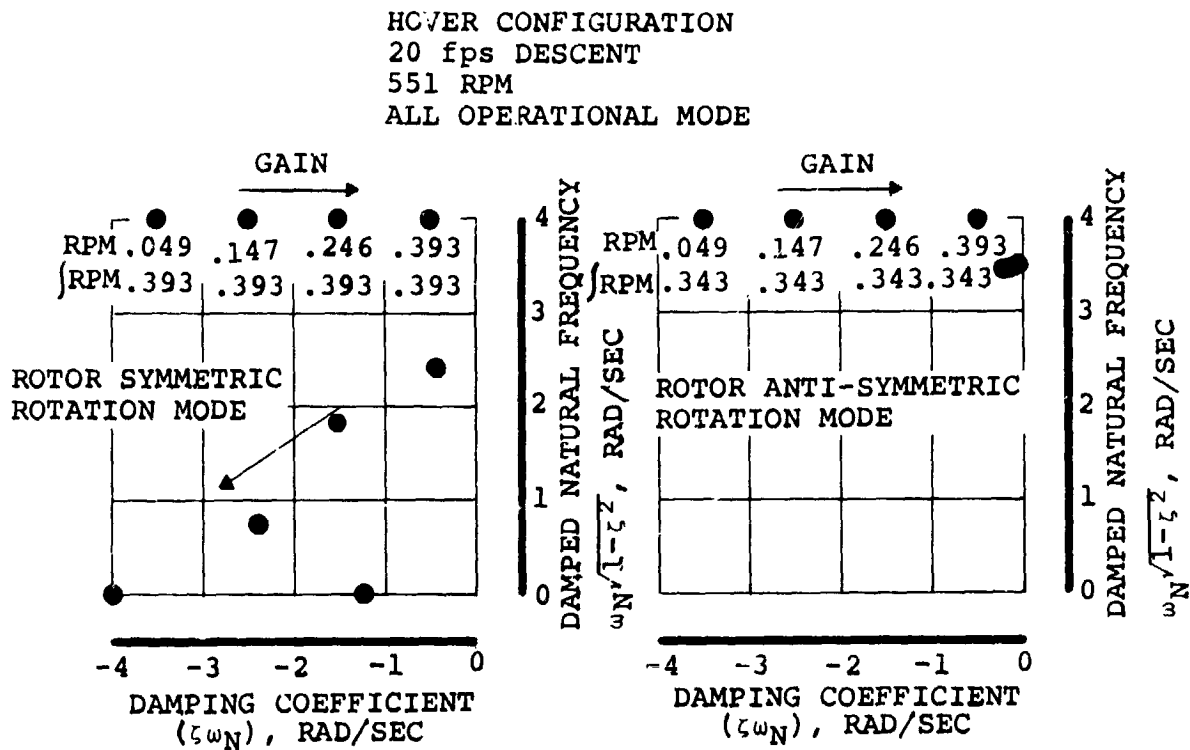
NOTES:

1. THIS GOVERNOR HAS NO EFFECT
   ON THE CROSS SHAFT (ANTI-
   SYMMETRIC ROTATION) MODE.
2. NO MODES ARE DRIVEN UNSTABLE
   WITH THE ABOVE GOVERNOR GAINS
3. SENSOR AND ACTUATOR DYNAMICS
   ARE INLCUDED IN THE MODEL.

FIGURE 7.19b. SINGLE GOVERNOR WITH TWO SENSORS ROTOR ROTATION
ROOT MIGRATION WITH $\int$ + RPM FEEDBACK

234

CRUISE CONFIGURATION
100 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN DEG/RPM

0.0 .109 .218 .327

CRUISE CONFIGURATION
200 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN ~ DEG/RPM

0.0 .054 .109 .163

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

DAMPED NATURAL FREQUENCY $\omega_N\sqrt{1-\zeta^2}$, RAD/SEC

CRUISE CONFIGURATION
300 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN ~ DEG/RPM

0.0 .021 .043 .065

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

NOTES:

1. ● SYMMETRIC ROTATION MODE
   O ANTI-SYMMETRIC ROTATION MODE
2. THIS GOVERNOR CONFIGURATION HAS NO EFFECT ON THE ANTI-SYMMETRIC ROTATION MODE
3. NO MODES ARE DRIVEN UNSTABLE WITH THE ABOVE GOVERNOR GAINS
4. SENSOR AND ACTUATOR DYNAMICS ARE INCLUDED IN THE MODEL
5. THE FOLLOWING UNCOUPLED WING MODES ARE INCLUDED IN THE MODEL
   o VERTICAL BENDING
   o CHORD BENDING
   o TORSION

FIGURE 7.20. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION WITH RPM FEEDBACK

235

CRUISE CONFIGURATION
100 KNOTS
386 RPM
ALL OPERATIONAL MODE

GAIN ~ RPM ⟶

| 0.0 | .109 | .218 | .327 |

DAMPED NATURAL FREQUENCY
$\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
200 KNOTS
386 RPM
ALL OPERATIONAL MODE

GAIN ~ RPM ⟶

| 0.0 | .0546 | .109 | .163 |

DAMPED NATURAL FREQUENCY
$\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
300 KNOTS
386 RPM
ALL OPERATIONAL MODE

GAIN ~ RPM ⟶

| 0.0 | .0218 | .0436 | .0655 |

DAMPED NATURAL FREQUENCY
$\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

NOTES:

1. ● SYMMETRIC ROTATION MODE
   ○ ANTI-SYMMETRIC ROTATION MODE
2. NO MODES ARE DRIVEN UNSTABLE
   WITH THE ABOVE GOVERNOR GAINS
3. SENSOR AND ACTUATOR DYNAMICS
   ARE INCLUDED IN THE MODEL
4. THE FOLLOWING UNCOUPLED WING
   MODES ARE INCLUDED IN THE MODEL
     ○ VERTICAL BENDING
     ○ CHORD BENDING
     ○ TORSION

FIGURE 7.21 SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION WITH RPM FEEDBACK

236

CRUISE CONFIGURATION
100 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN ~ DEG/SEC/RPM

0.0 .1635 .218 .327

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

20

10

0

-20   -10   0
DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
200 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN ~ DEG/SEC/RPM

0.0 .1365 .273 .409

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

20

10

0

-20   -10   0
DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
300 KNOTS
386 RPM
CROSS SHAFT FAILURE MODE
GAIN ~ DEG/SEC/RPM

0.0 .1365 .273 .355

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

20

10

0

-20   -10   0
DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

NOTES:

1. ● SYMMETRIC ROTATION MODE
   O ANTI-SYMMETRIC ROTATION MODE
2. THIS GOVERNOR CONFIGURATION HAS
   NO EFFECT ON THE ANTI-SYMMETRIC
   ROTATION MODE
3. NO MODES ARE DRIVEN UNSTABLE
   WITH THE ABOVE GOVERNOR GAINS
4. SENSOR AND ACTUATOR DYNAMICS
   ARE INCLUDED IN THE MODEL
5. THE FOLLOWING UNCOUPLED WING
   MODES ARE INCLUDED IN THE MODEL
      o VERTICAL BENDING
      o CHORD BENDING
      o TORSION

FIGURE 7.22. SINGLE GOVERNOR WITH TWO SENSORS
             ROTOR ROTATION ROOT MIGRATION WITH INTEGRAL FEEDBACK

CRUISE CONFIGURATION
100 KNOTS
386 RPM
ALL OPERATIONAL MODE
GAIN →

0.0   .273   .327   .382

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
200 KNOTS
386 RPM
ALL OPERATIONAL MODE
GAIN →

0.0   .1365   .273   .355

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

CRUISE CONFIGURATION
300 KNOTS
386 RPM
ALL OPERATIONAL MODE
GAIN →

0.0   .1365   .273   .355

DAMPED NATURAL FREQUENCY $\omega_N \sqrt{1-\zeta^2}$, RAD/SEC

-40   -30   -20   -10   0

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC
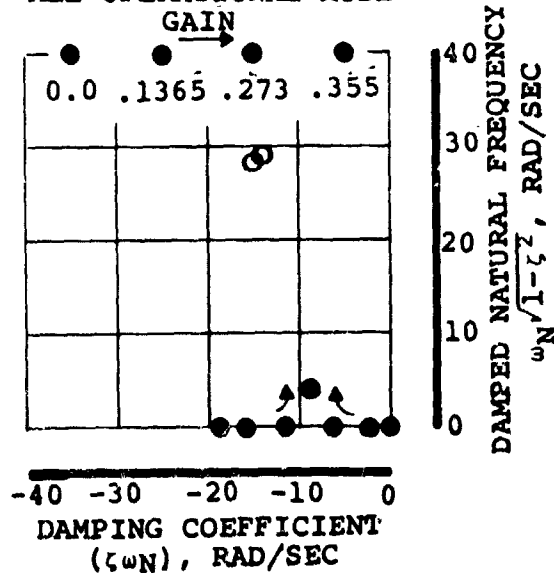
NOTES:
1. ● SYMMETRIC ROTATION MODE
    O ANTI-SYMMETRIC ROTATION MODE
2. THIS GOVERNOR CONFIGURATION HAS
    NO EFFECT ON THE ANTI-SYMMETRIC
    ROTATION MODE
3. NO MODES ARE DRIVEN UNSTABLE
    WITH THE ABOVE GOVERNOR GAINS
4. SENSOR AND ACTUATOR DYNAMICS
    ARE INCLUDED IN THE MODEL
5. THE FOLLOWING UNCOUPLED WING
    MODES ARE INCLUDED IN THE MODEL
      o VERTICAL BENDING
      o CHORD BENDING
      o TORSION

FIGURE 7.23. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION
WITH ∫RPM FEEDBACK

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Flight Condition	Integral of RPM Feedback Gain (Steady State Gain)	RPM Feedback Gain (Steady State Gain)
Cruise - 100 Kts	.327 deg/sec/RPM	.0164 deg/RPM
Cruise - 200 Kts	.273 deg/sec/RPM	0.0
Cruise - 300 Kts	.355 deg/sec/RPM	0.0
Hover - Steady	.475 deg/sec/RPM	.268 deg/RPM
Hover - 20 fps Climb	.306 deg/sec/RPM	.175 deg/RPM
Hover - 20 fps Descent	.393 deg/sec/RPM	.246 deg/RPM

In the hover regime the range of steady state gains is not so critical with climb or descent airspeed and it would be acceptable to have a constant gain in the hover regime of:

O .475 deg/sec/RPM

O .245 deg/RPM (Figure 7.24)

In the cruise regime, the minimum requirements can be satisfied throughout the flight envelope with steady state gains of:

O .355 deg/sec/RPM

O .0177 deg/RPM (Figure 7.24)

7.5 EFFECT OF GOVERNOR ON FLYING QUALITIES

In the hover configuration, the governor system may influence the vertical translation and roll modes. Since the chosen governor configuration averages RPM from each rotor, there will be no effect when the aircraft rolls since one rotor will

2-3
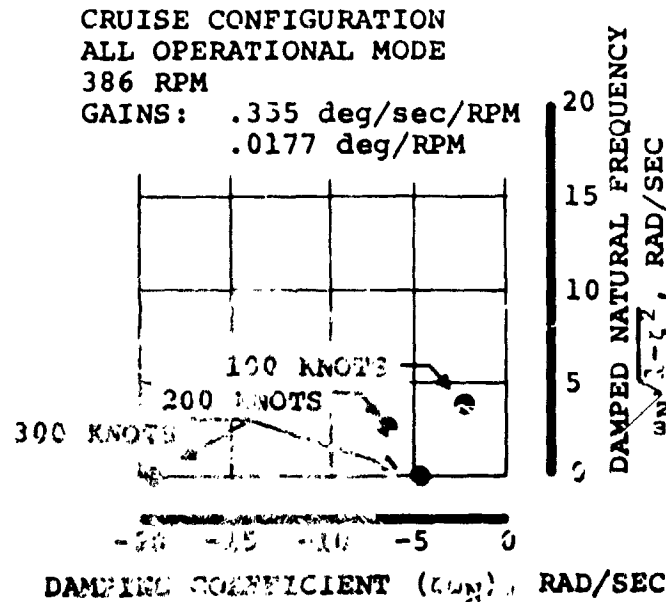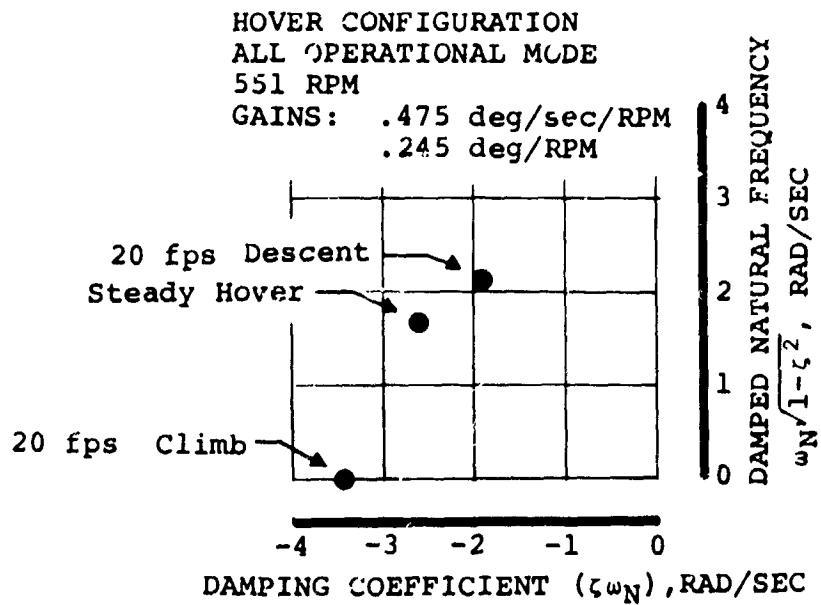
FIGURE 7.24. SINGLE GOVERNOR WITH TWO SENSORS
ROTOR ROTATION ROOT MIGRATION WITH VELOCITY

tend to increase RPM and the other will tend to decrease RPM. In the vertical mode both rotors tend to move in unison. The governor will tend to maintain constant thrust and thus decrease vertical damping. With no governor the RPM would tend to stay constant for a short period of time (because of rotor inertia effects) and the thrust would increase or decrease - whichever would be in the stable sense - during this interval. Figure 7.25 illustrates the governor effect on the roll and vertical modes in root locus form for a tilt rotor aircraft in the hover configuration. Note that for the vertical mode the root is moved toward the right half plane when the governor is operational which indicates a less stable condition. In fact the vertical damping is reduced by approximately 50%, but as depicted by Figure 7.26 this is still an acceptable damping level.

In the cruise configuration the dutch roll and phugoid modes may be affected. The dutch roll mode will be unaffected by the single governor with two sensors for the same reasons that the roll mode in hover is unaffected (Figure 7.27). The phugoid mode will be adversely affected by the governor because the governor reacts very fast when compared to the phugoid period thus effectively eliminating any prop/rotor effect (T_u AND M_u). For the high wing configuration the rotor effect is stabilizing therefore loss of the prop/rotor effect will destabilize the phugoid (i.e., the aircraft will behave as a jet in this mode). There is an interesting trade-off in the cruise configuration between phugoid damping and aircraft gust

HOVER CONFIGURATION
STEADY HOVER
551 RPM
ALL OPERATIONAL MODE

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps DESCENT
551 RPM
ALL OPERATIONAL MODE

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

HOVER CONFIGURATION
20 fps CLIMB
551 RPM
ALL OPERATIONAL MODE

DAMPING COEFFICIENT
$(\zeta\omega_N)$, RAD/SEC

NOTES:

O 1. NO GOVERNOR
△ 2. COLLECTIVE PITCH
      GOVERNOR
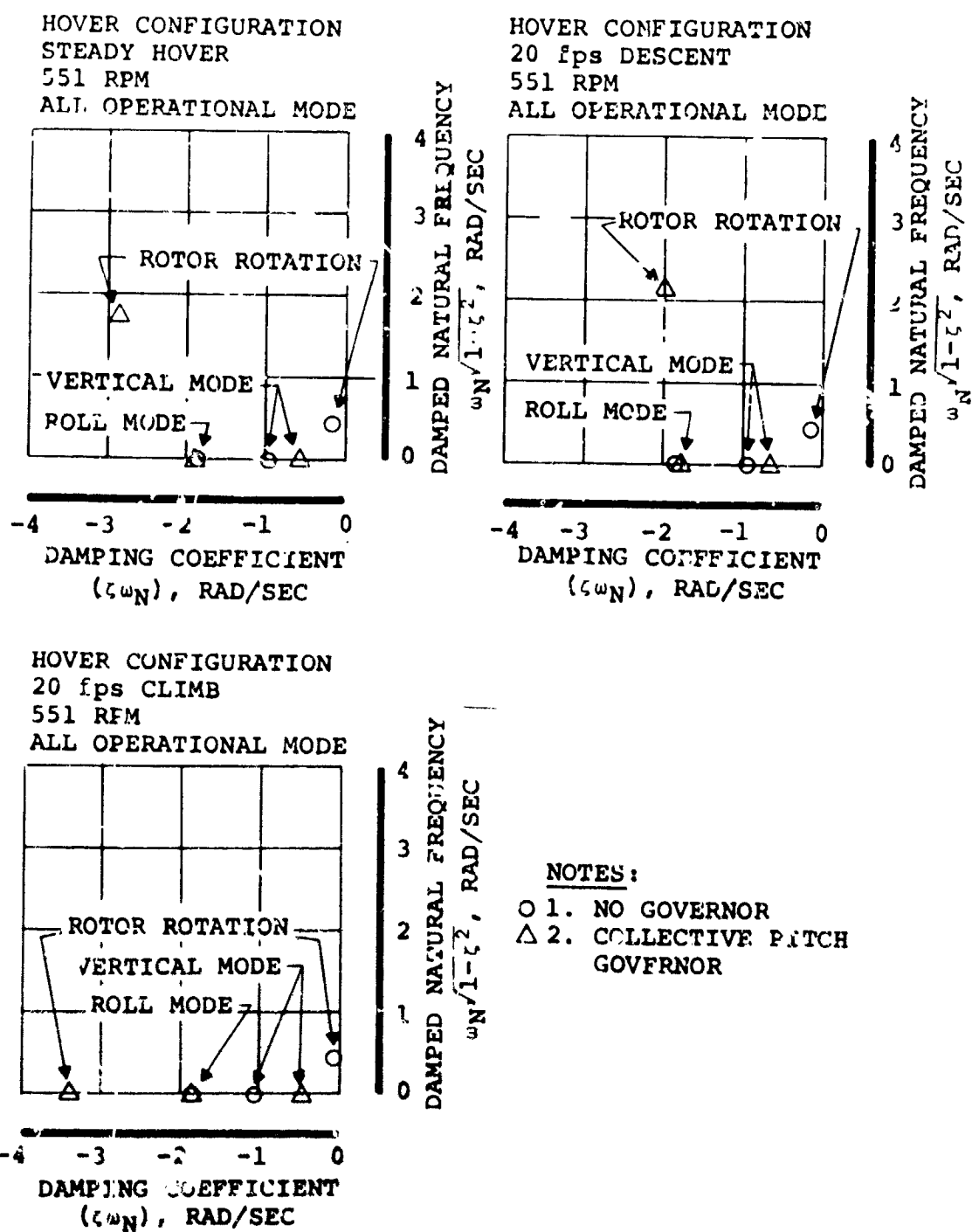
FIGURE 7.25.   EFFECT OF GOVERNOR ON HOVER ROLL AND
               VERTICAL TRANSLATION MODES

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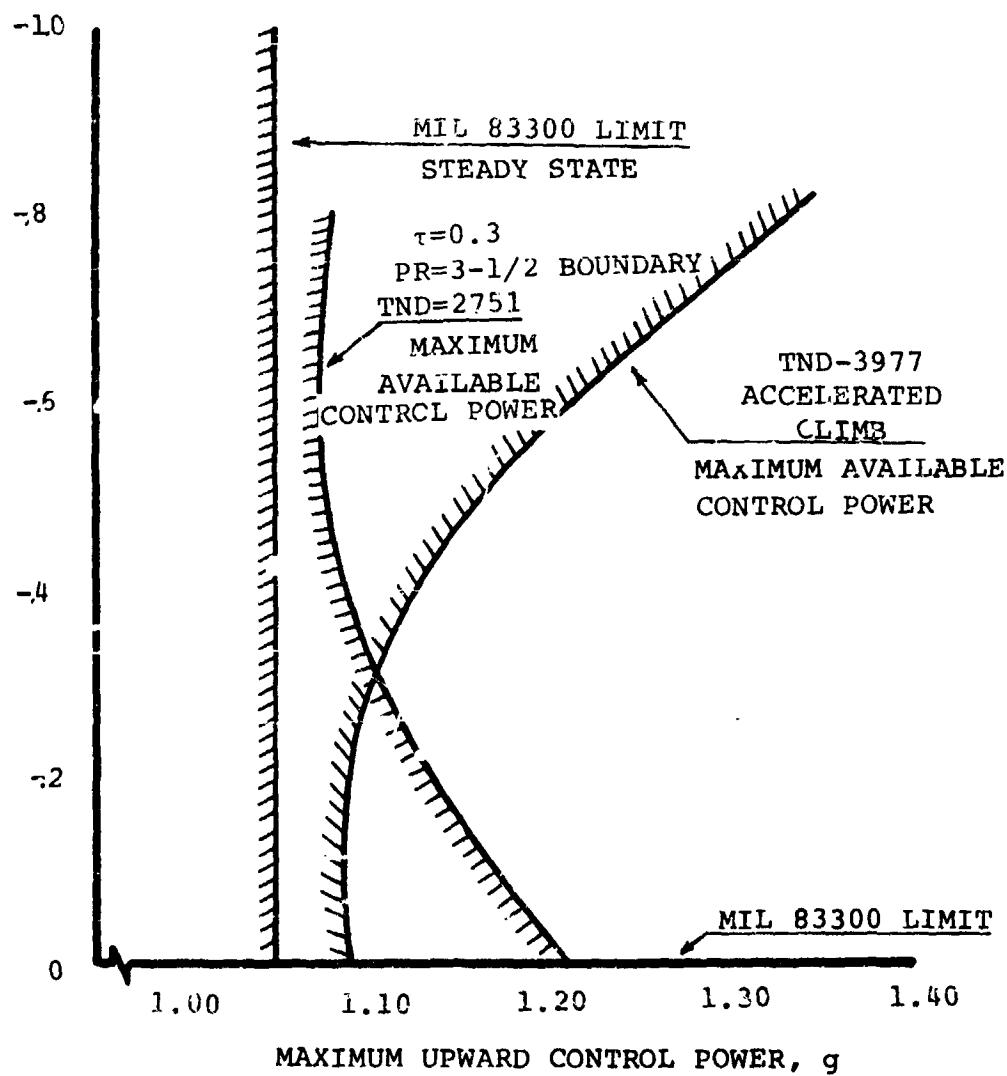
FIGURE 7.26. CRITERIA FOR ACCEPTABLE HOVER CONTROL

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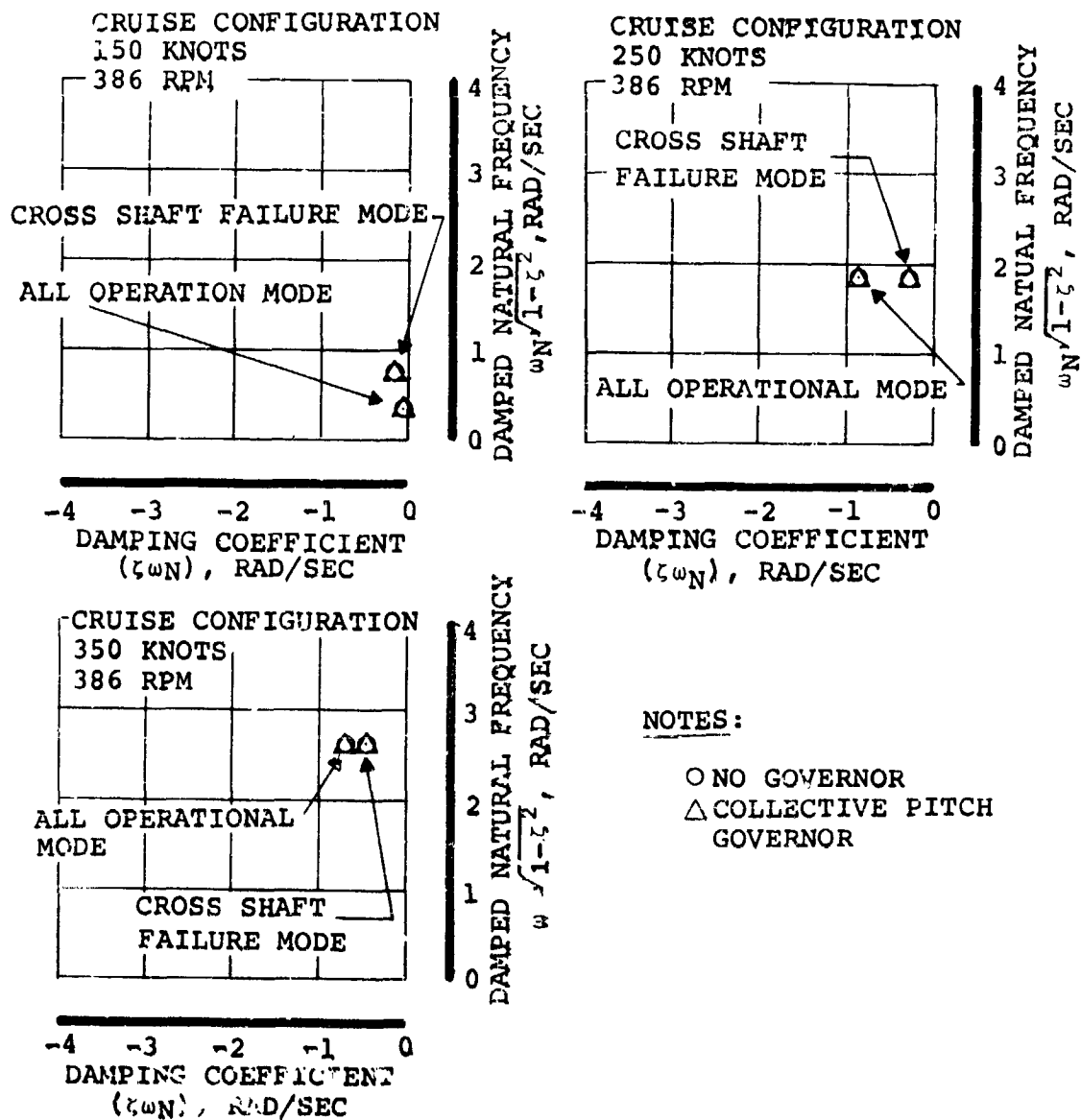
FIGURE 7.27. EFFECT OF GOVERNOR ON DUTCH ROLL MODE

response. For a very "clean" aircraft it may be necessary to have some auxiliary damping from the prop/rotors to meet the MIL Spec 8785B level 1 requirements of a damping ratio of at least .04 in the phugoid mode. One way to obtain this would be to reduce the governor efficiency (and axial gust response relief) to retain some phugoid damping from the rotors. However, in early tilt rotor configurations one of the most frequently expressed pilot comments concerned the aircraft axial gust response (AFFTC-TR-60-4), therefore it seems for future tilt rotor configurations, aircraft gust response will necessarily be heavily weighed and the trade-off between gust response and phugoid damping cannot be considered lightly.

7.6 GUST RESPONSE

On earlier tilt rotor and tilt wing aircraft, which were designed more from a helicopter viewpoint than fixed-wing, one recurring pilot comment was the poor turbulence response along the longitudinal axis (AFFTC-TR-60-4). Since that time it has become apparent that at least part of the adverse response problem stems from the use of fuel governing rather than collective pitch governing. Figure 7.28 illustrates in a crude manner three different modes of RPM control and the effect of each when hit by an axial gust. There are five parameters to be considered when evaluating axial gust response:

O gust input

O RPM

O fuel flow
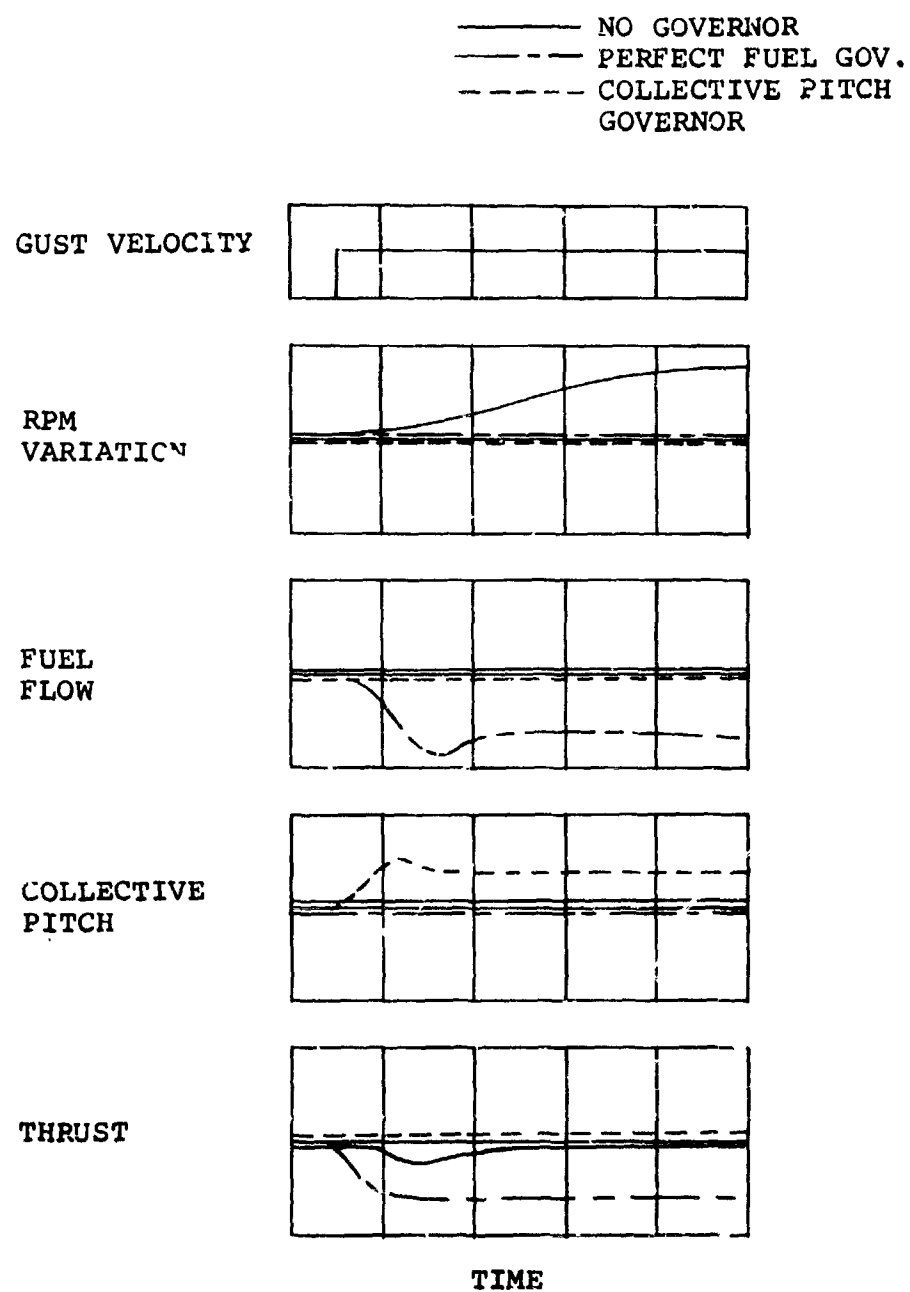
FIGURE 7.28. EFFECT OF GOVERNOR SYSTEM CONFIGURATION
ON POWER SYSTEM PARAMETERS

O   collective pitch

O   thrust

Of the three control modes , consider first the case of no RPM
governing. When hit by an axial gust, fuel flow and collec-
tive pitch stay constant and rotor RPM starts to vary. The
result is that in the steady state the thrust does not change
substantially from its pre-gust value. For a high inertia
rotor there will be some change in thrust as the rotor inertia
delays the rpm decay whereas for a small inertia rotor or
propeller the RPM will nearly track the gust shape and little
thrust variation will be seen. From a gust response point
of view, a low rotational inertia rotor having no governor is
an acceptable solution; however, without a governor the RPM
variations are of such magnitude as to make the system unaccept-
able. Next consider the perfect fuel governor (i.e., one that
keeps RPM absolutely constant by controlling fuel flow). When
the rotor is hit by a gust the fuel flow increases or de-
creases, collective pitch remains constant and RPM is constant
resulting in a change in thrust. The change in thrust will
result in acceleration along the longitudinal axis which will
be detrimental to the ride and handling qualities of the air-
craft. The fuel governor then is the converse of the no go-
vernor case - RPM control is acceptable but gust response is
not. The collective pitch governor varies rotor blade angle
to control RPM at constant power setting. When the rotor is
hit by a gust, fuel flow remains unchanged and collective

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pitch varies in a manner to maintain constant RPM and constant thrust. This configuration governor then solves the aircraft axial gust problem by:

O reducing aircraft gust response

O eliminating the RPM variation

Time history gust responses were run for the three modes of operation discussed above. Collective pitch governor dynamics and gains were employed as defined in paragraph 7.4.3. Perfect fuel governing was evaluated to establish an upper limit gust response for that configuration and to show a comparison with collective pitch style governing. The no governor case allowed the rotor to freely spin on the hub. Response in both hover and cruise configurations was evaluated. RPM variation, cabin acceleration and significant structural responses have been plotted for each case.

Two gust shapes were used in the evaluation: 1-cos and random. The 1-cos shape was used to evaluate the governor response in cruise and hover (i.e., determine if the accuracy and transient response goals were met). Random turbulence was used in the 300 knot cruise case to illustrate the aircraft response with governor operative when encountering turbulent air. The turbulence model was obtained from NASA and is described in Reference 7.2. Results of the gust response study were much as expected except that the adverse acceleration response of the fuel governor in cruise was worse than anticipated. Figures

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7.29 through 7.31 show the results.

In the cruise configuration, the collective pitch governor effectively keeps the RPM within the 2% transient error and the .3% steady state error criteria while reducing the longitudinal acceleration response to gusts to a near zero level. Significant structural blade elastic modes show no adverse response. In the worst case (fuel governing) the structural load and blade modal response is well below allowable limits. Comparison of the fuel governor and collective pitch governor traces indicate at a glance the desirability of the collective pitch governor from a gust response viewpoint. Longitudinal acceleration is reduced from a peak value of 9 ft/sec^2 to 2 ft/sec^2 at 100 knots (the worst case for the collective pitch governor) and at 300 knots from 7 ft/sec^2 to less than 1 ft/sec^2. Thus at cruise velocities the pilot may fly through a 20 fps axial gust with minimum aircraft response. Use of the collective pitch governor should effectively solve the longitudinal acceleration problem noted in AFFTC-TR-60-4. Improvement in the structural gust response is also noted with collective pitch governing. Although even with fuel governor the blade modal response and wing chord bending response is small, the collective pitch governor improves the response on the same order of magnitude as the acceleration response is improved. Figure 7.32 illustrates the aircraft flying through clear air
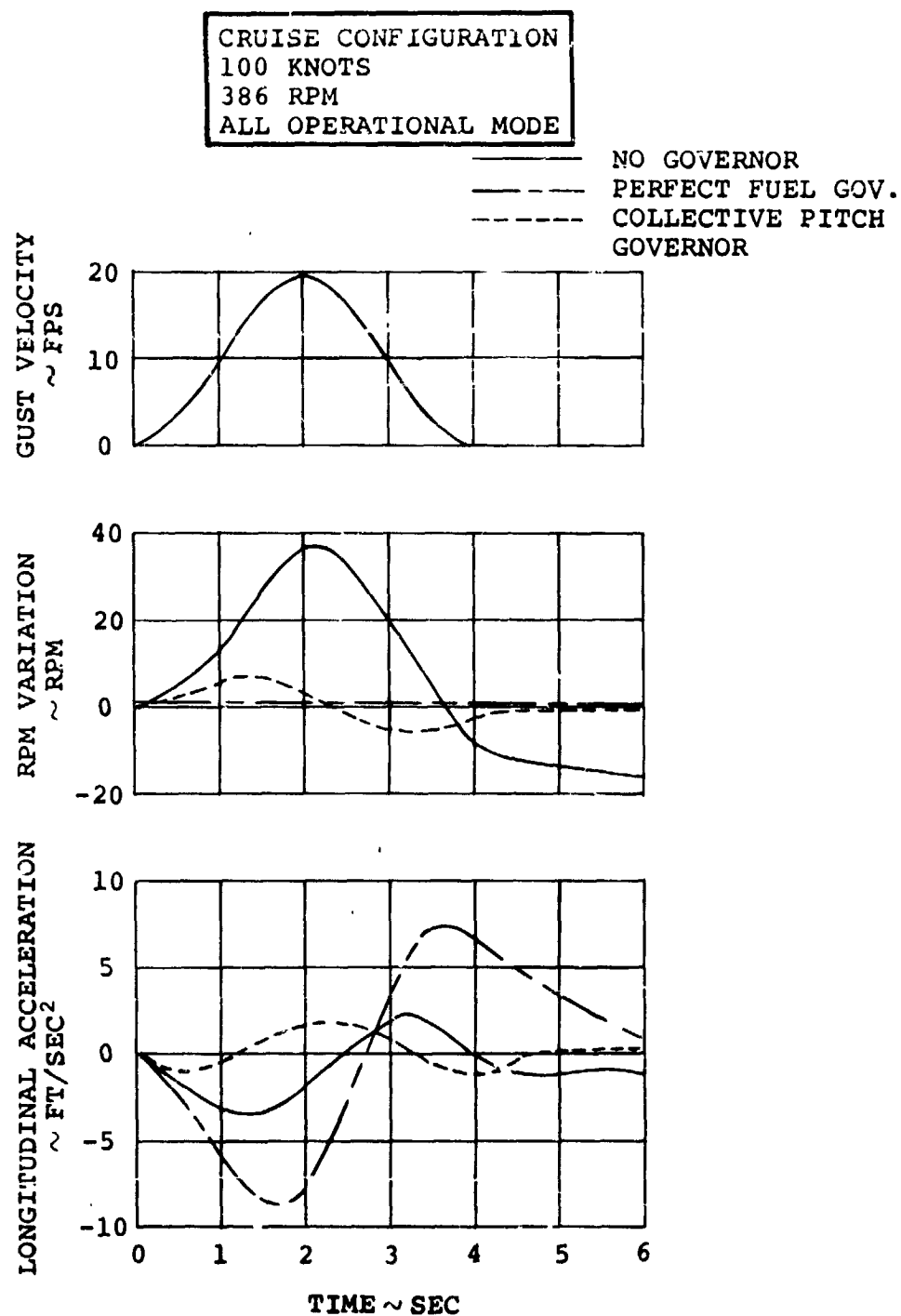
FIGURE 7.29a.   AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
100 KNOT CRUISE

250

CRUISE CONFIGURATION
100 KNOTS
386 RPM
ALL OPERATIONAL MODE

——————— NO GOVERNOR
——— ——— PERFECT FUEL GOV.
——————— COLLECTIVE PITCH
GOVERNOR
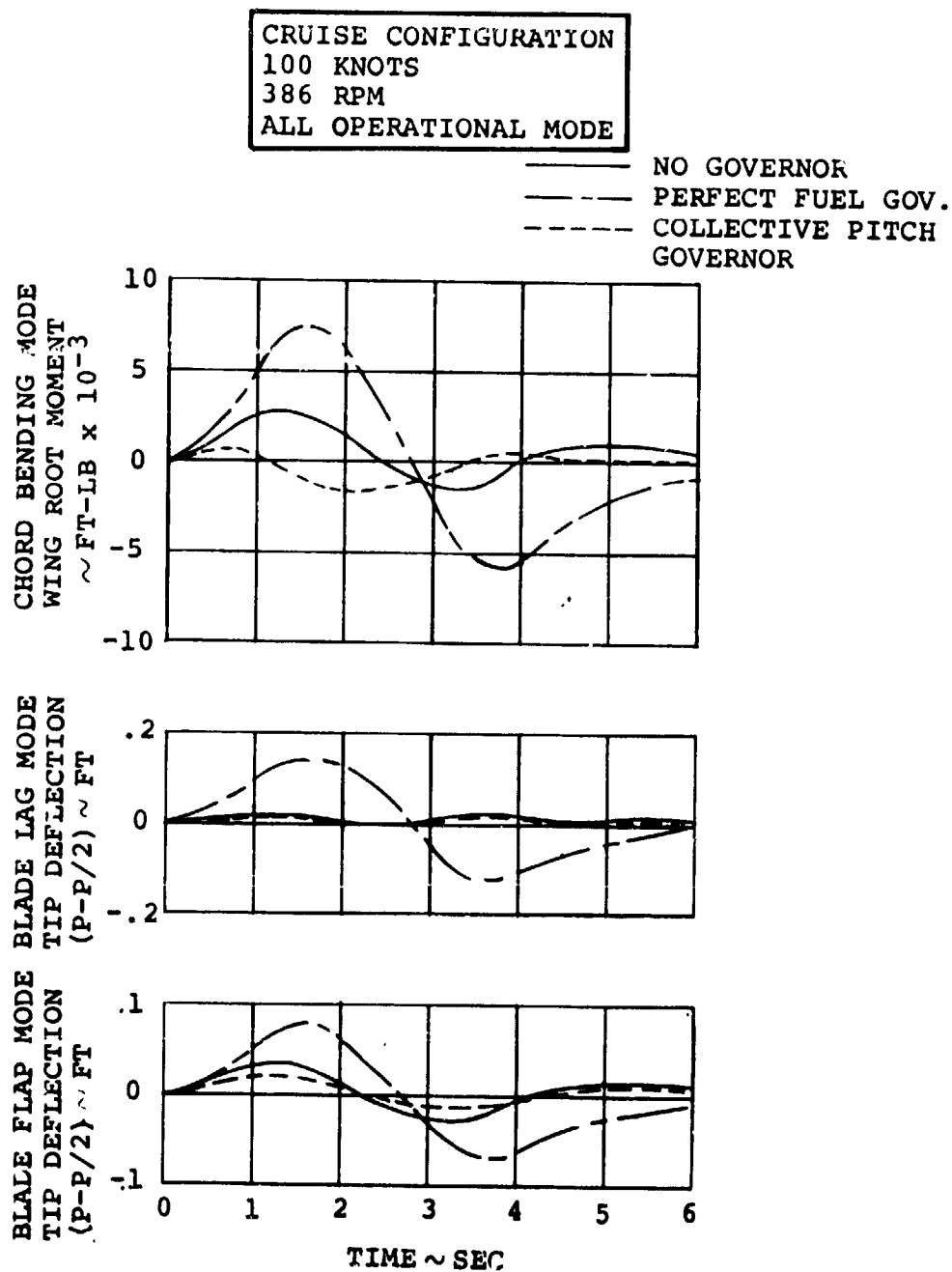
FIGURE 7.29b.   AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
100 KNOT CRUISE

251
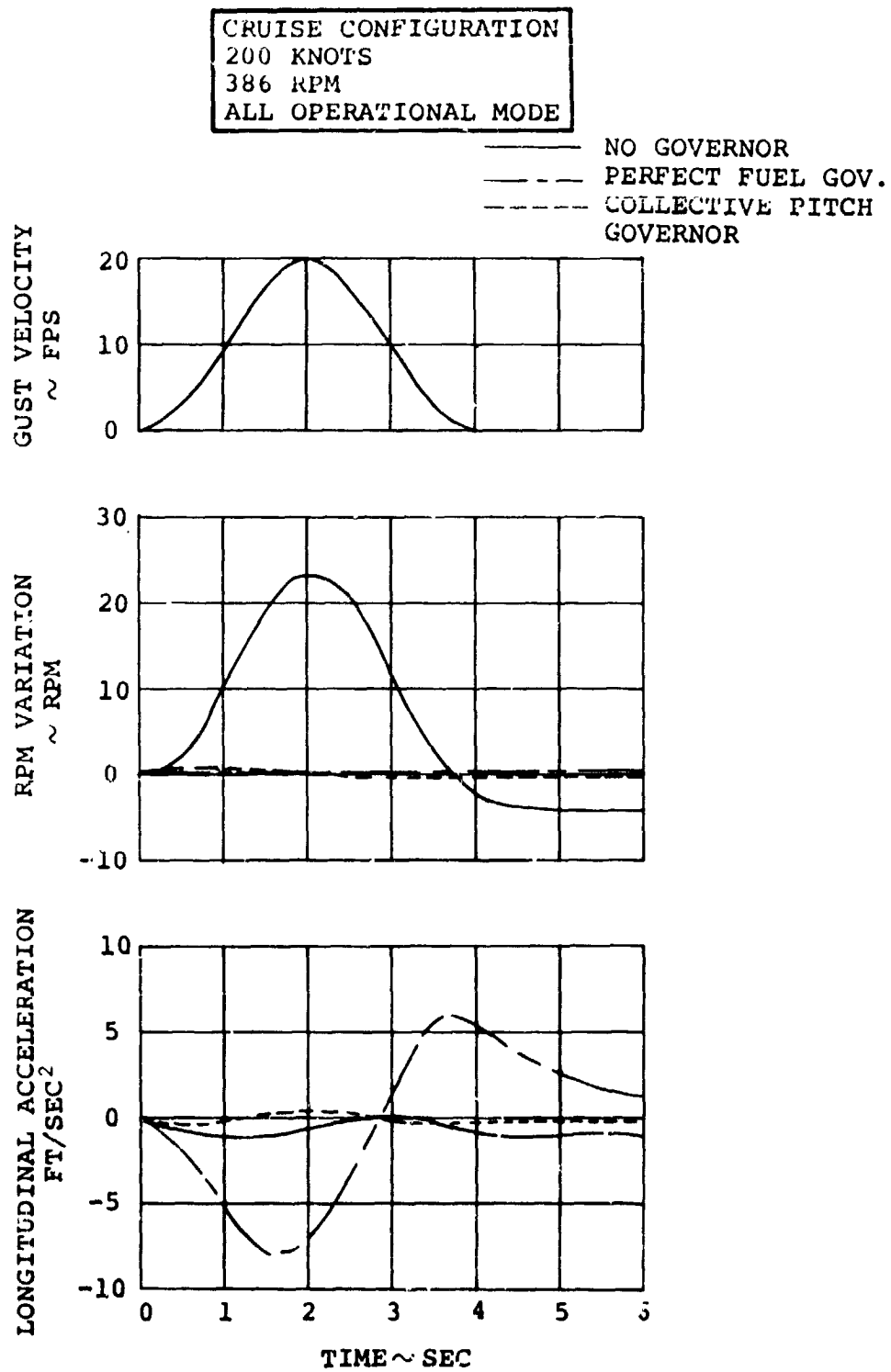
FIGURE 7.30a. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
200 KNOT CRUISE
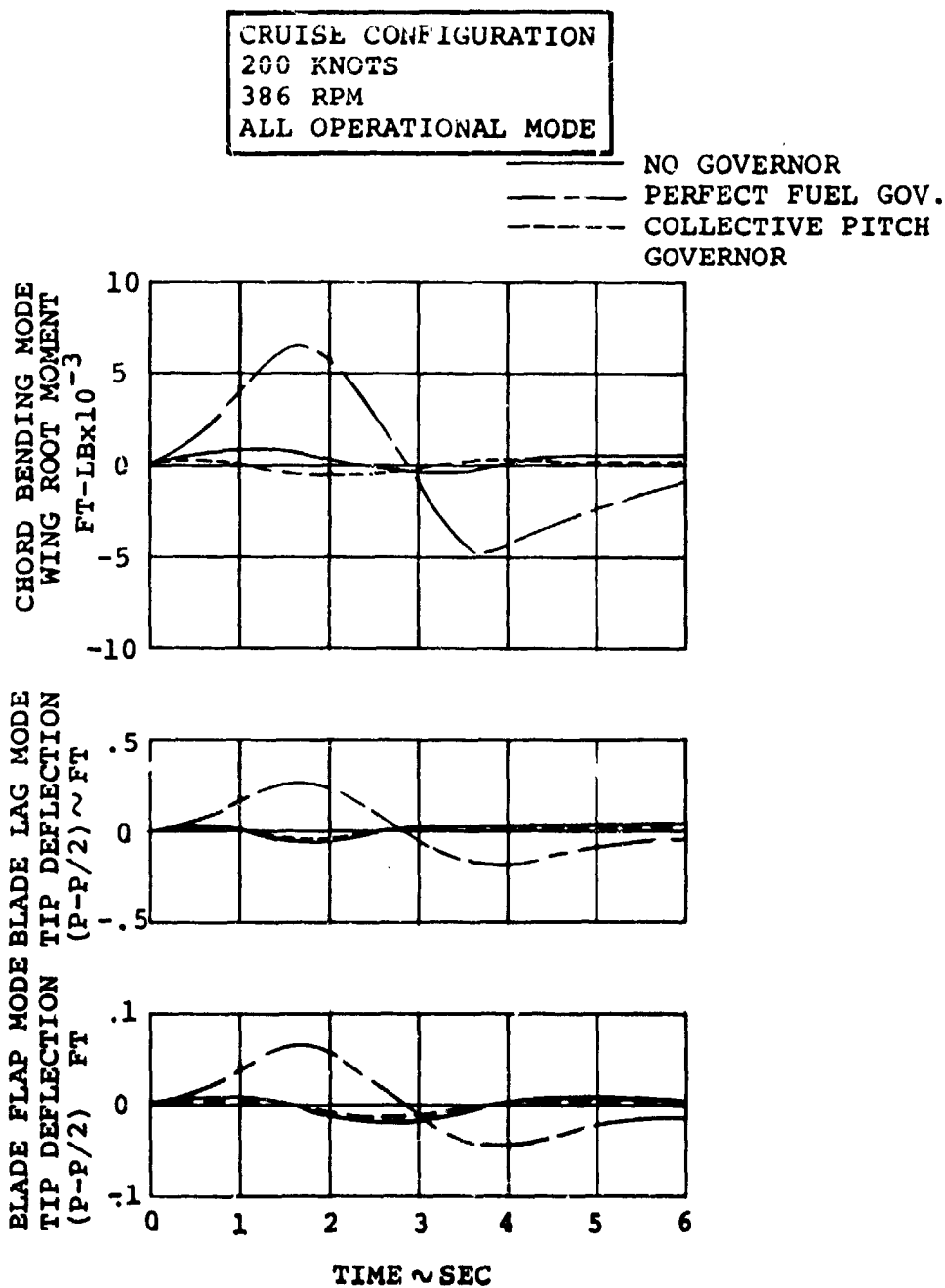
FIGURE 7.30b. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
200 KNOT CRUISE
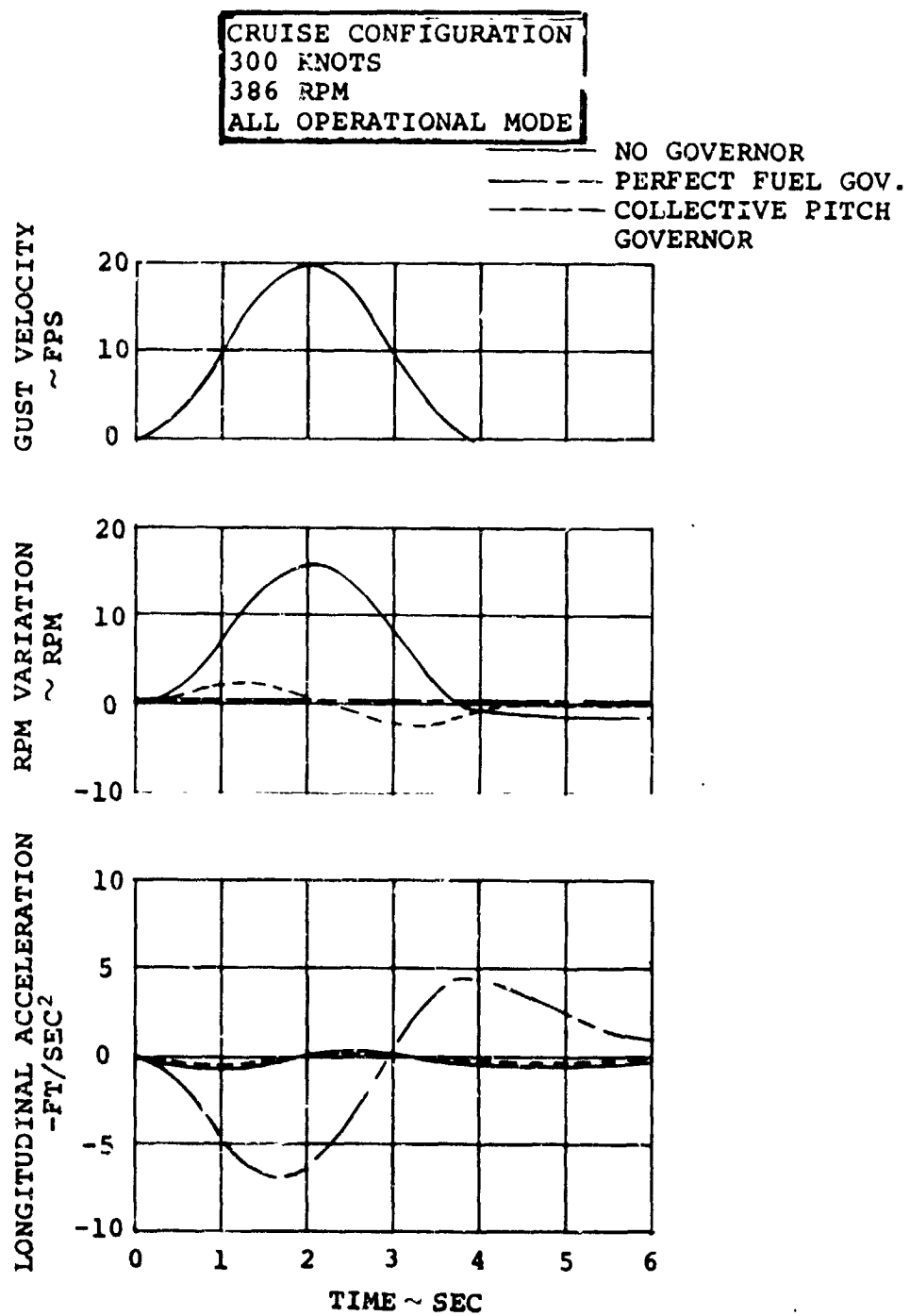
FIGURE 7.31a. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST,
300 KNOT CRUISE

254
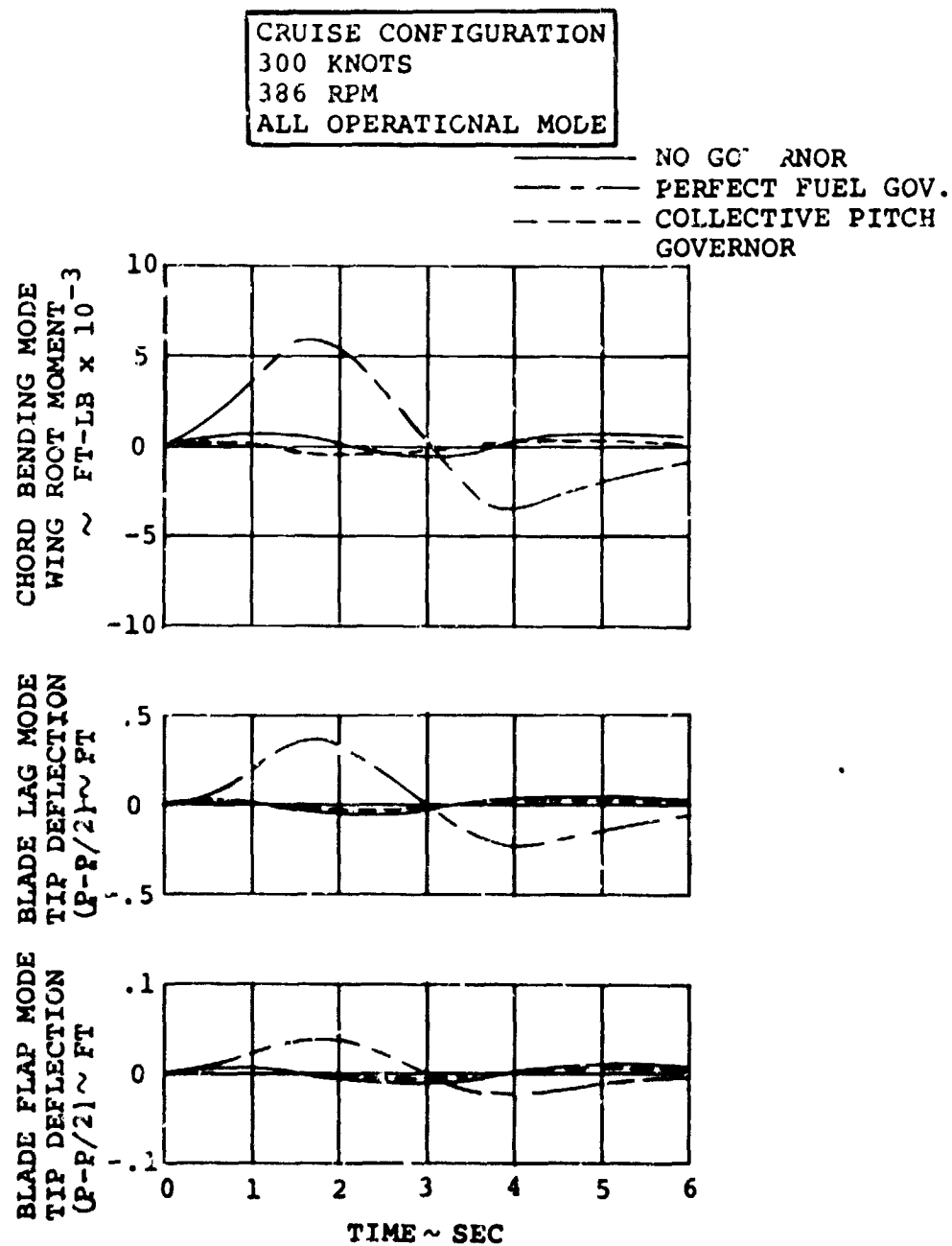
FIGURE 7.31b. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
300 KNOT CRUISE

255
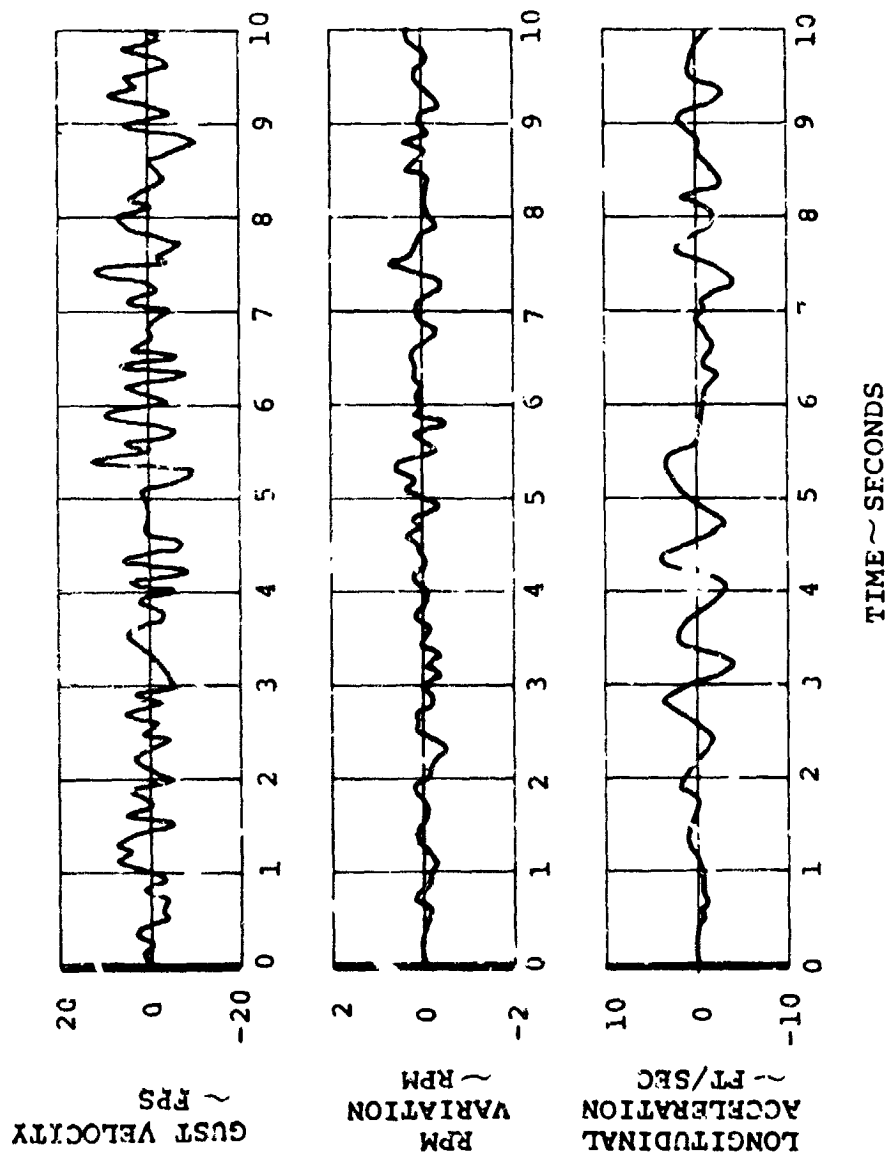
FIGURE 7.32. AIRCRAFT RESPONSE TO AXIAL RANDOM TURBULENCE
300 KNOTS - COLLECTIVE PITCH GOVERNOR

turbulence at 300 knots with the governor operative. Note that the same desirable characteristics are present as observed with the 1-cos gust. The RPM error is well within the specified bounds and in fact the peak variation is never greater than 1 RPM (.3% error). Longitudinal acceleration is attenuated with maximum acceleration never greater than .125 "g".

In the hover configuration, collective pitch governing improves the aircraft gust response in much the same manner as cruise, even if the improvement in gust response is not as striking. The effect is shown in Figures 7.33 through 7.35. RPM variation is well within the transient and steady state goals, acceleration and structural response is improved. Once again the wing bending moments are well within allowable limits for an aircraft of this type and the blade responses are small enough to be considered insignificant.

One major difference between cruise and hover is the similarity in vertical acceleration for the no governor and fuel governor cases. Because of the slower rotor response (lower natural frequency) than in cruise, the rotor does not respond as quickly and there is a larger vertical acceleration response. Although good gust response is desirable in hover, the RPM hold capability is more important along with the effect on flying qualities as described in paragraph 7.5.

The collective pitch governor has a significant beneficial effect on aircraft gust response in both cruise and hover. In

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fact the analysis shows that in cruise the collective pitch
governor eliminates one portion of the adverse gust response
seen in earlier tilt rotor aircraft.

7.7 CONCLUSIONS

A collective pitch governor which senses the RPM variation of
each rotor and feeds back collective pitch proportionally to
the averaged signal has been chosen as the configuration which
satisfies the requirements of a tilt rotor aircraft. RPM
variation has been shown to be within the error criteria for
1-cos and random axial gusts. In the rigid body modes which
may be affected there is either no degradation or a small de-
gradation which is not harmful to the flying qualities of
the aircraft. The most significant improvement this governor
configuration makes on the aircraft is in axial gust response
in cruise. The "bucking" or "jerking" of earlier tilt rotor
configurations has been eliminated so that the aircraft may
now fly in turbulent conditions with a minimum of longitudinal
response in cruise.
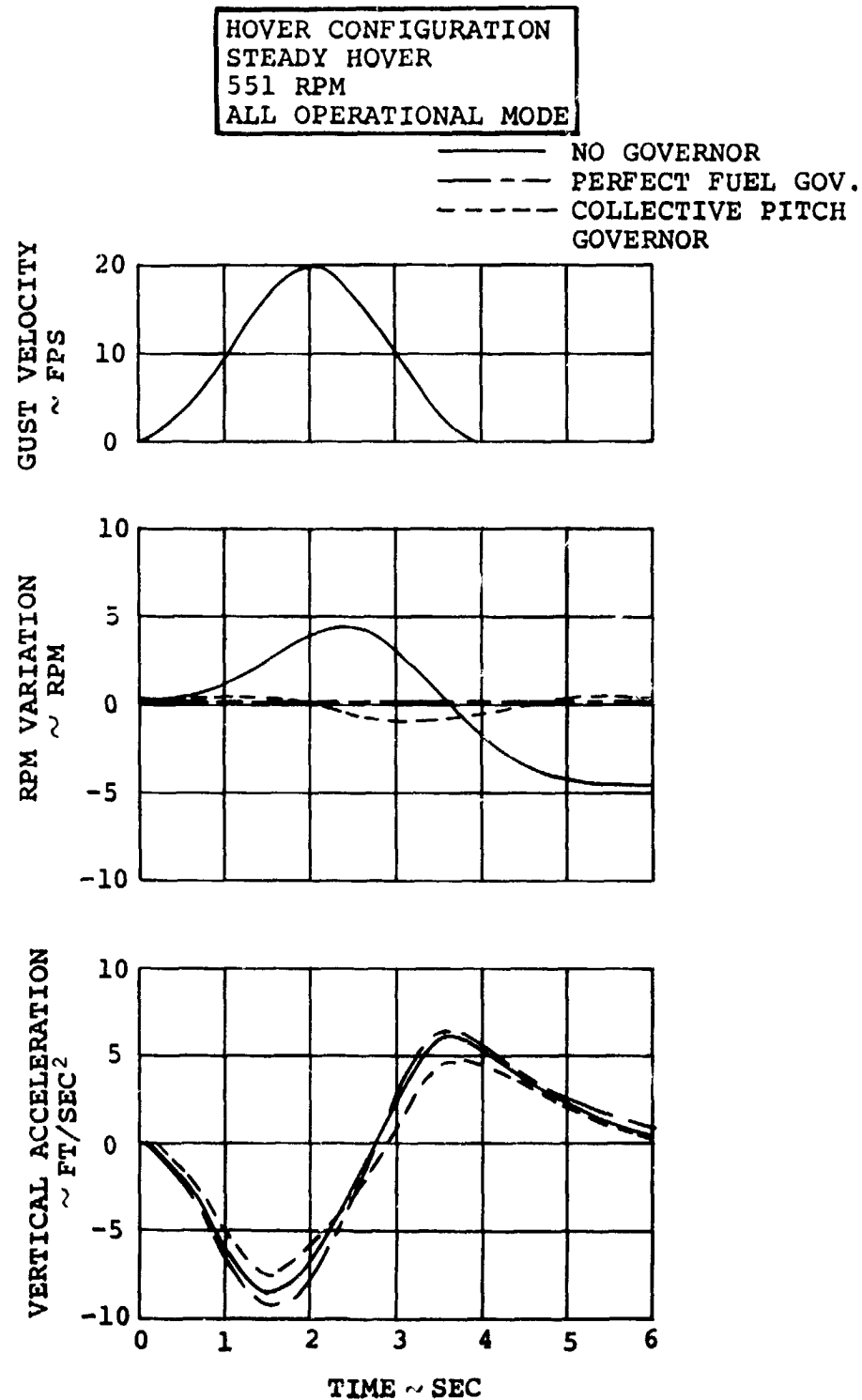
FIGURE 7.33a. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
STEADY HOVER
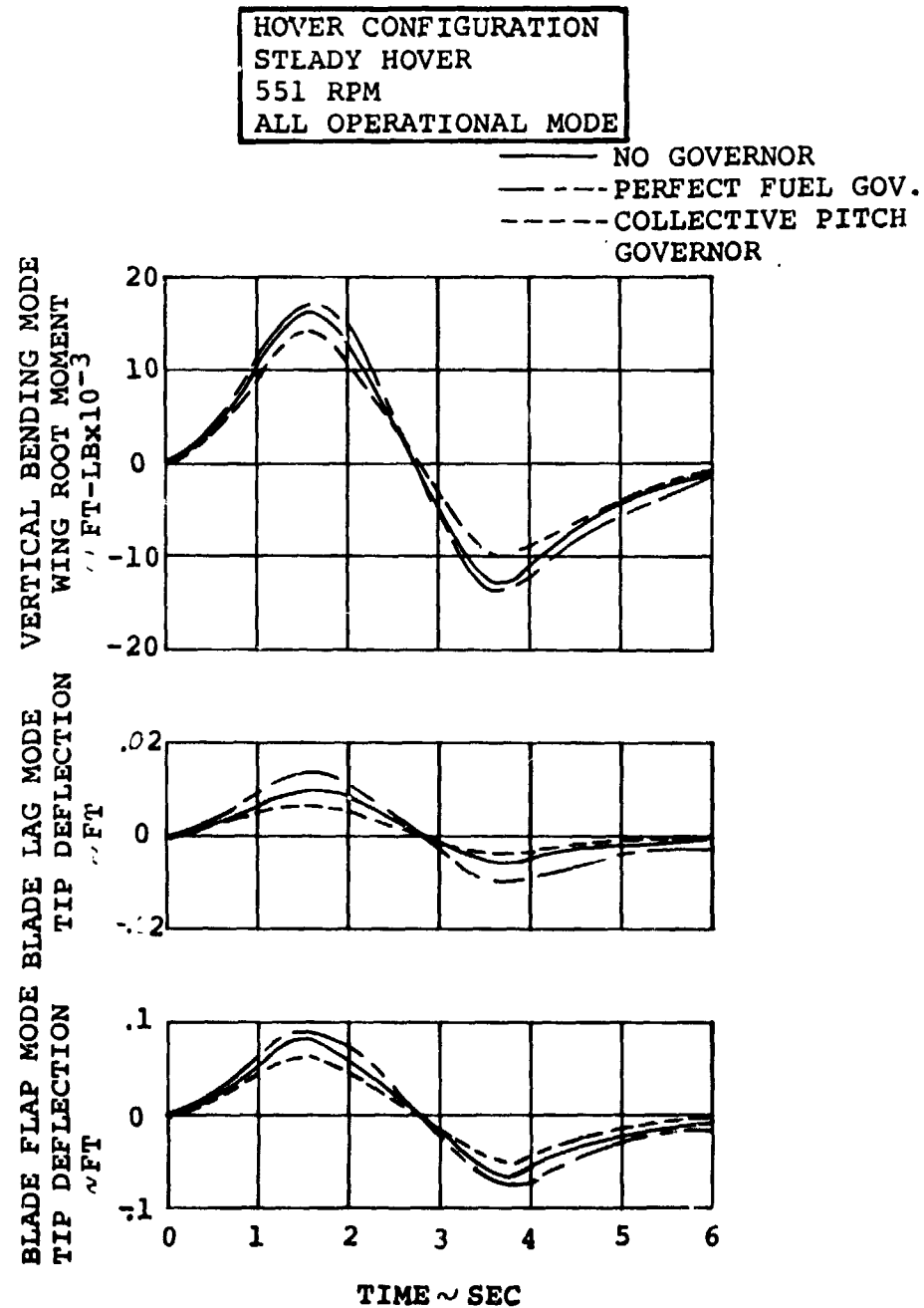
FIGURE 7.33b. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
STEADY HOVER
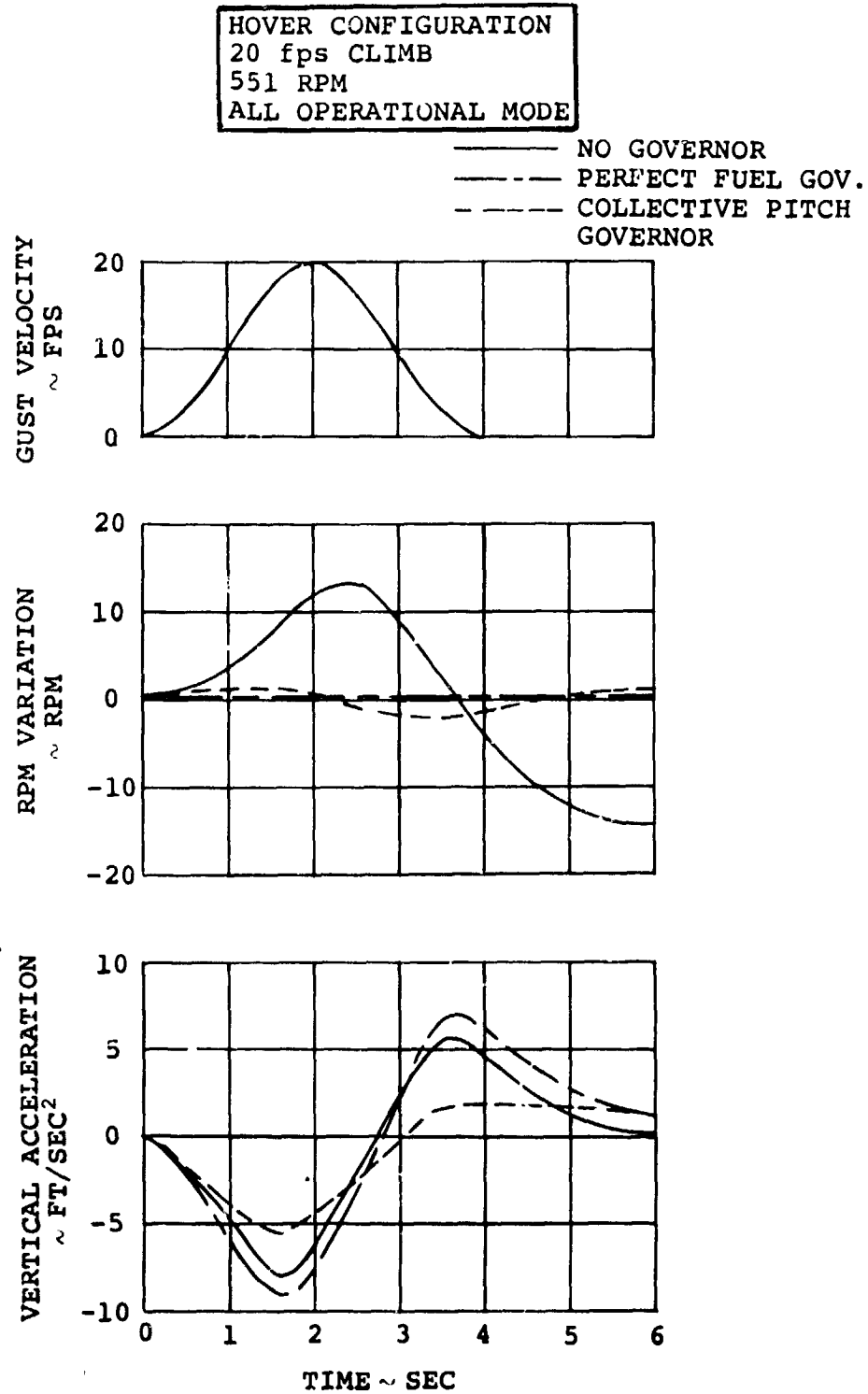
FIGURE 7.34a. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
HOVER - 20 fps CLIMB

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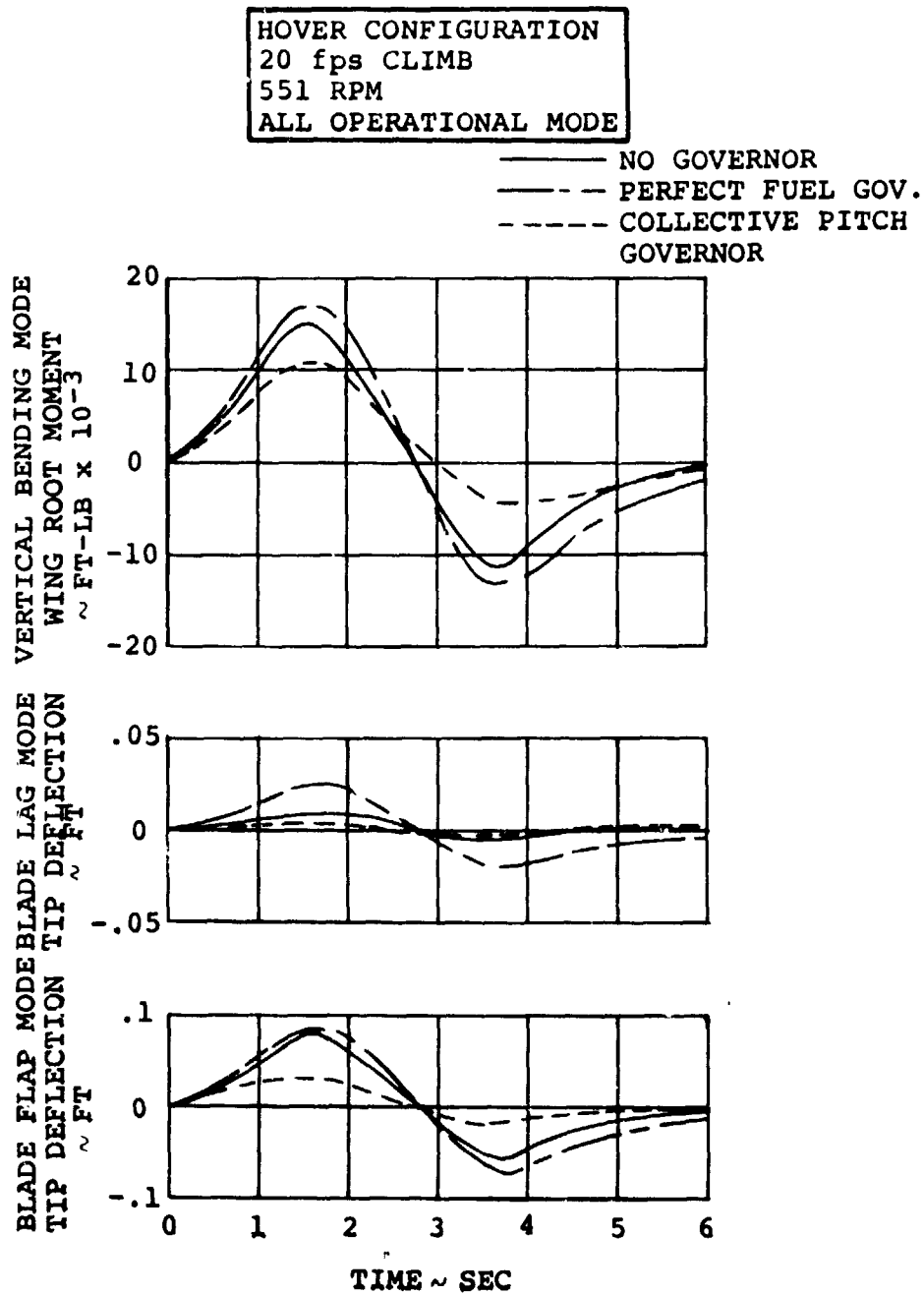
FIGURE 7.34b. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
HOVER - 20 fps CLIMB
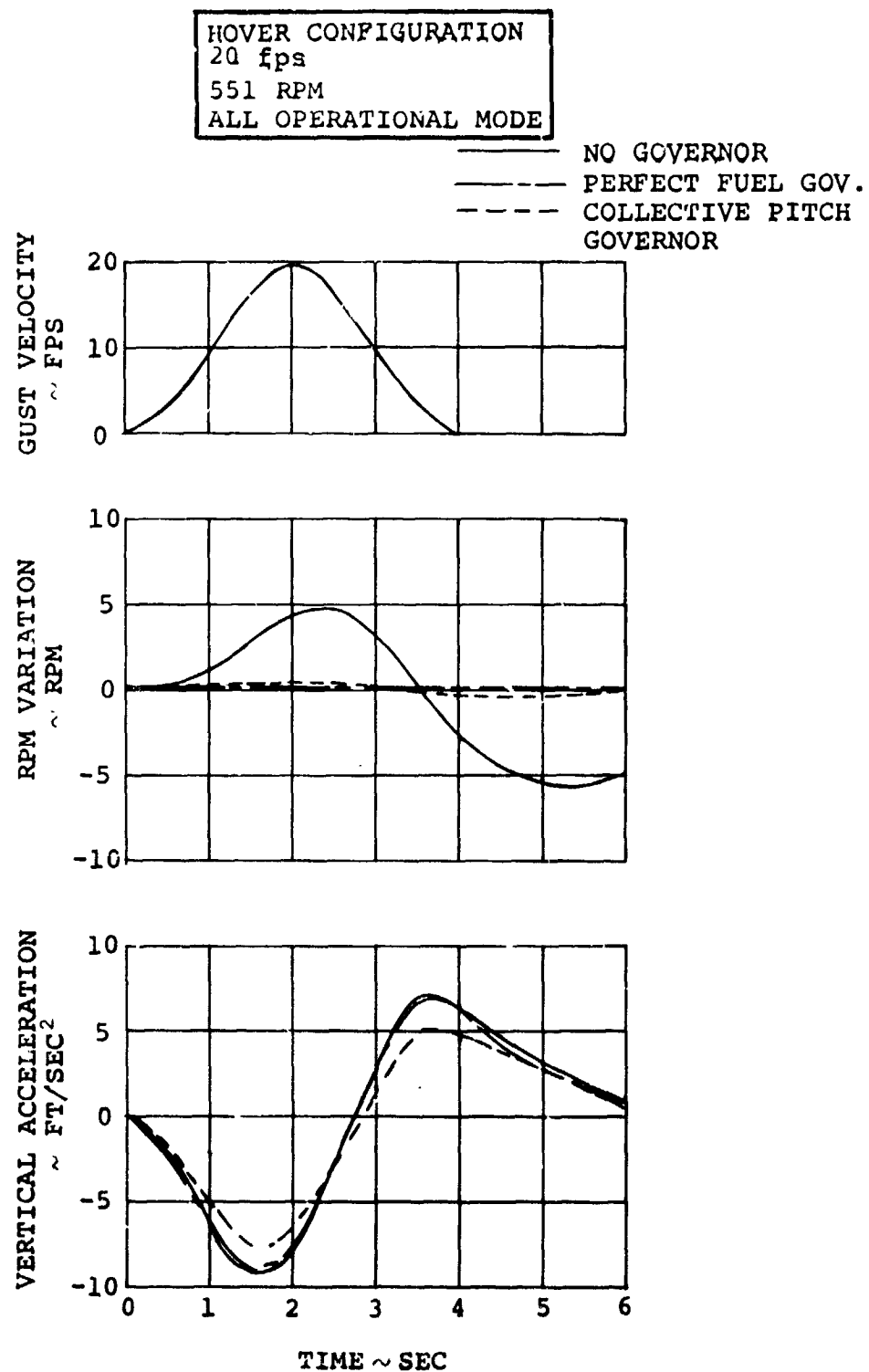
FIGURE 7.35a. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
HOVER - 20 fps DESCENT
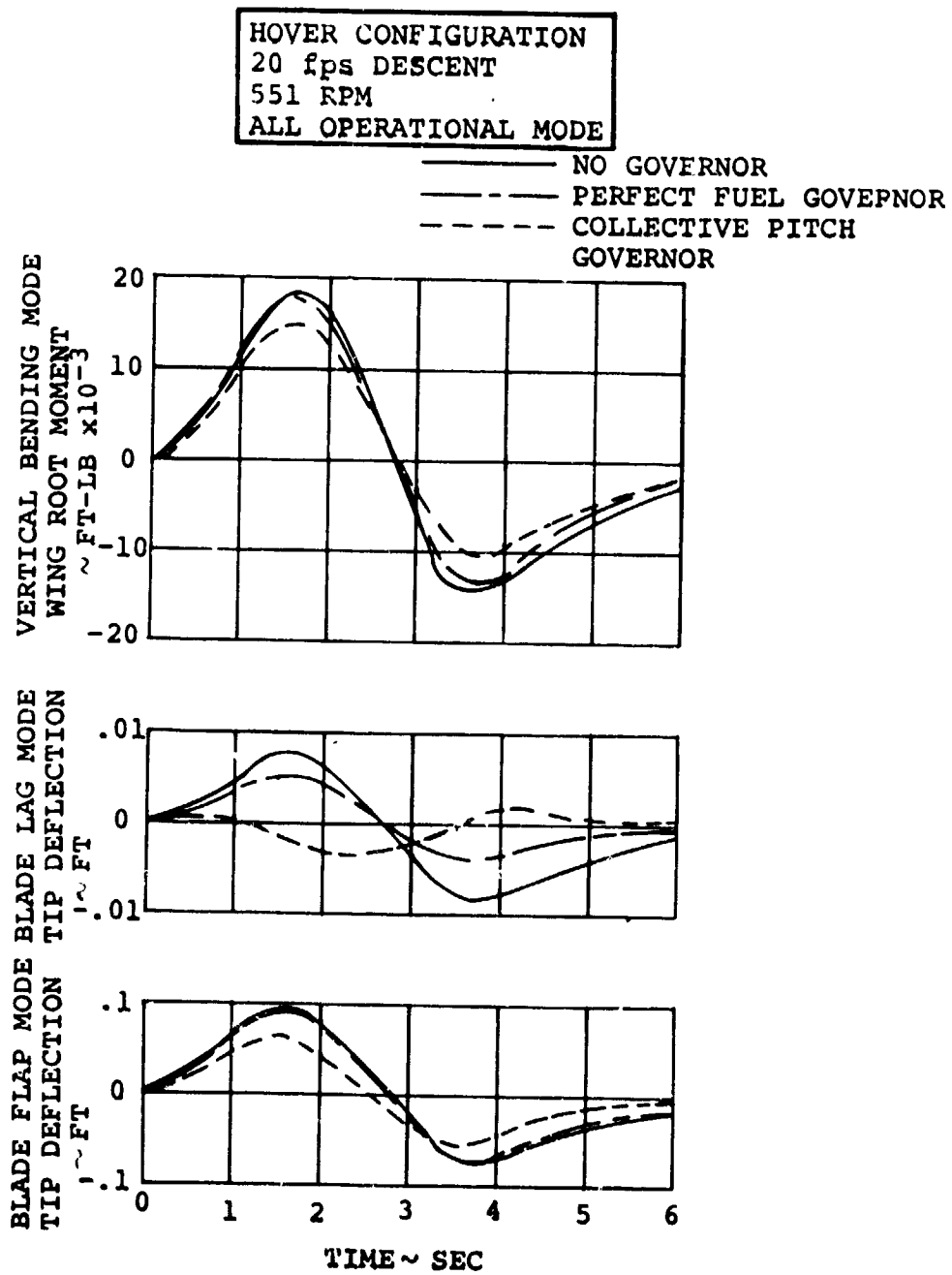
FIGURE 7.35b. AIRCRAFT RESPONSE TO 1-COS AXIAL GUST
HOVER - 20 fps DESCENT

8. APPLICATIONS OF SWASHPLATE FEEDBACK SYSTEMS TO FOLDING

TILT ROTOR CONFIGURATIONS

Applications of swashplate feedback for the folding tilt rotor
aircraft fall into two categories:

1. loiter or low speed cruise, and

2. spin up and spin down.

In both of the above categories the level of blade loads is
not prohibitive without feedback; however, the loads may be
reduced to a lower level for gust alleviation or stability
augmentation.

In the loiter or low speed cruise regime the same concepts of
load alleviation may be applied as in Paragraphs 3 and 7. The
rotor will be fully operational and spinning at design RPM;
hence, the configuration will be identical to tilt rotor
configurations previously analyzed. An area of concern for
this system will be, if and how, to phase out swashplate feed-
backas the aircraft approaches the folding regime of flight.
This problem will not be one of advancing the technology, but
rather one of gain scheduling.

Spin up and spin down of the rotor may be a possible cause of
high blade loads. To investigate the level of these loads, a
scale model test of the folding tilt rotor was undertaken in
1971 using a 1/9 scale semi-span model. The results may be
reviewed in Ref. 8.1, Air Force Technical Report AFFDL-TR-71-62,
Vol. VII. In summary, various spin up and spin down schedules
were tried to investigate the effect on blade loads. In each

case the loads were small. There was not a large variation in loads for the different spin schedules and, in fact, the final schedule was chosen on a basis of drag, not blade loads. It therefore appears that feedback will not be required during spin up and spin down, and it may be convenient to phase it out at some rpm lower than normal cruise.

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9. CONCLUSIONS AND RECOMMENDATIONS

The analytical studies reported in Sections 2 through 8 confirm that substantial benefits may result from the proper use of feedback using the   controls and also the conventional controls.   i a high level of confidence in the validity of the re. .   sed on the correlation with test data per.o.med using the methodology of this report. Examples of this predictive and correlation capability are given in the text and a more extensive coverage of its topic is given in the test report on the full scale test of the Model 222 26-foot diameter rotor in the NASA-Ames 40 X 80-foot tunnel. Hence the broad conclusions and recommendations of the report may be accepted without reservation.

The parametric studies on rotor derivatives show that the low in-plane stiffness rotor enjoys a negative hub pitching moment under normal flight conditions so that the net destabilizing effect of the rotor on static stability is less than that which would exist for a high stiffness blade and this is confirmed by test data. This has beneficial effects on horizontal stabilizer area requirements.

Studies in blade load alleviation and stability augmentation show that various alternative objectives may be met using cyclic feedback. Cases examined included minimization of blade flapping, and minimization of net pitching moment about the aircraft center of gravity. No stability problems were en-

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countered although additional shaping was recommended in
one case to increase the phase margin. It was found that the
requirements of gain and azimuth might be different in dynamic
gust response cases compared with steady state alleviation
and methods of resolving these requirements are identified
and recommended for further study.

The use of feedback through the conventional control systems
for direct lift control and alleviation of normal acceleration
response in turbulent air was studied and found to be effective
in cruise and less effective in transition. The reductions
in response demonstrated were 45% in cruise and 15% in transi-
tion. These results are extremely encouraging since they
were obtained without special effort at optimization, or
inclusion of elevator feedback effects to trim the pitching
moments due to flap and spoiler application.

Collective pitch governor studies indicated that a governing
system which senses the RPM variation of each rotor and feeds
back collective pitch proportional to the averaged signal
should be chosen as the configuration which satisfies the
requirements of a tilt rotor aircraft. RPM variation has been
shown to be within the error criteria for (1-COS) and random
axial gusts. This governor system does not produce any signi-
ficant degradation in the flying qualities of the aircraft and
the response to axial or head on gusts is very substantially
attenuated.

Extensive work on moda` suppression showed that very sub-
stantial increases in damping of selected · des could be
accomplished and this result has been fully confirmed by
full scale tests in the 40 X 80-foot tunnel. Attempts to
introduce additional damping to the blade modes in hover
showed some promise, but this topic was not fully developed
because it is of somewhat academic interest at the present
time.

A general conclusion which emerges from these studies is that
the full potential of some systems depends on interaction
with an associated system. These relationships are summarized
in Table 9.1. Thus, the effectiveness of a blade load
alleviation system will be enhanced if aircraft acceleration
response to gusts is reduced by direct lift control. Con-
versely the effectiveness of a direct lift feedback system
may be improved when working in conjuction with a high rate
swashplate modal suppression system. Thus, there is a
strong case for the eventual implementation of all the
feedback controls listed, however, priority should be given
to the conventional SAS, rotor governing and blade load
alleviation systems.

TABLE 9.1. SUMMARY OF FEEDBACK CONTROL SYSTEM RELATIVE APPLICATIONS AND INTERACTIONS

ITEM	SENSORS SIGNALS	BENEFITS	LIMITATIONS	RELATION TO OTHER SYSTEMS
BLADE LOAD ALLEVIATION/ STABILITY AUGMENTATION SYSTEM	AQ OR STRAIN	REDUCTION IN BLADE LOADS AND IMPROVE-MENT IN FLYING QUALITIES.	BLADE LOADS NOT COM-PLETELY CONTROLLED BY CYCLIC FEEDBACK. A/C ACCELERATIONS ALSO FORCE BLADES.	BLADE LOAD CONTROL CAN BE IMPROVED FURTHER BY CON-CURRENT USE OF DIRECT LIFT FEEDBACK.
DIRECT LIFT	AQ OR NORMAL ACCELERATION	ATTENUATES ACCEL-ERATIONS DUE TO GUSTS. IMPROVES HEIGHT CONTROL.	ATTENUATION PROBABLY LIMITED BY WING MODAL RESPONSE, WHICH MAY BE EXCITED BY SYSTEM. MAY REQUIRE ASSIST FROM MODAL ATTENUATION SYSTEM.	WILL HELP BLADE LOAD SYSTEM BY REDUCING ACCELERATION FORCING OF THE BLADES.
ROTOR COLLECTIVE/ GOVERNING	RPM	CONTROLS RPM AND BLADE RESPONSE TO AXIAL GUSTS. ALSO FORE AND AFT ACC-ELERATION IN CRUISE. DOES NOT DEGRADE A/C LATERAL DIRECTIONAL RESPONSE TO GUSTS.	EFFECTIVENESS NOT SIGNIFI-CANTLY LIMITED IN CRUISE OR HOVER SINCE LONGITUDINAL FORCES ARE PREDOMINANTLY FROM ROTOR. SOME LIMITA-TION IN NORMAL "G" ATTENU-ATION AT HIGH SPEED END OF TRANSITION DUE TO WING LIFT.	NO SIGNIFICANT INTERACTION WITH OTHER SYSTEMS.
MODAL SUPPRESSION SYSTEM	MODAL ACCELER-ATION OR RATE	DAMPING INCREASE IN AEROELASTIC MODES. MAY BE USED TO AUG-MENT EFFECTIVENESS OF DIRECT LIFT "G" CONTROL.	EFFECTIVENESS REDUCED AT HIGH FREQUENCIES DUE TO BLADE TIME CONSTANTS. SENSITIVE TO FLIGHT CONDITION.	PROBABLY HIGH LEVEL OF INTERACTION WITH DIRECT LIFT SYSTEM.
CONVENTIONAL SAS	A/C RATES, SIDE-SLIP AND ANGLE OF ATTACK	PROVIDES ACCEPTABLE SHORT PERIOD, ETC. CHARACTERISTICS.	DOES NOT ATTEMPT TO PRO-VIDE POSITIVE CONTROL OF GUST RESPONSE LOADS, ETC.	RELATIVELY LITTLE DIRECT INTERACTION. USES SAME ACT-UATORS ADDING OR SHARING AUTHORITY.

10. HARDWARE CONSIDERATIONS

10.1 INTRODUCTION

From the standpoint of hardware implementation the feedback control is subdivided into two major subsystems: the Stability Augmentation System (SAS) which accomplished the functions of rate damping, load alleviation, mode suppression, and turn coordination, and the Governor System which accomplished the function of rotor speed stabilization and collective control quickening in hover. In the SAS system we are primarily concerned with the load alleviation and modal suppression sections of the system. Changes in the currently proposed Model 222 hardware which would be needed to implement the recommendations of this report are discussed in paragraphs 10.4 and 10.5. In each case changes have been suggested which would reflect the minimum cost and modification to the Model 222 flight control system.

10.2 STABILITY AUGMENTATION SYSTEM (SAS)

SAS stabilization occurs through differential electrohydraulic actuators (extensible link type) inserted in the primary mechanical flight control system. The actuation points for the SAS are as follows primary longitudinal control for longitudinal rate damping, longitudinal load alleviation and longitudinal mode suppression; primary lateral control for lateral rate damping; primary directional control for yaw rate damping; lateral cyclic control at each rotor for lateral load alleviation and lateral mode suppression. A typical hardware functional block diagram of the SAS is

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shown in Figure 10.1.

10.2.1 Redundancy

The level of redundancy of the SAS is dictated by the required authority, and by the SAS failure requirements.

A fail functional system requires at least triple redundancy regardless of the authority. Fail functional is defined as the capability of the system to continue to perform its function without any degradation after any single failure.

A fail operative system requires at least dual redundancy. Fail operative is defined as the capability of the system to perform its function after any single failure with any or all of the following restrictions: a) Reduction of system authority after failure; b) Tolerable transient at failure; c) Manual shutdown of failed channel to restore system function. Reduction of failure transient in a fail operative system can be achieved through cross coupling of SAS feedbacks. A hard-over in one channel link is partially compensated by motion of the other link(s) in the opposite direction commanded by the cross feedback. If the required SAS authority is ample enough, then avoidance of an intolerable transient at first failure in a fail operative system may require a level of redundancy above dual. Because the required SAS authority can be significantly different from one control axis to another, different levels of SAS redundancy can be adopted for the various control axes.

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10.2.2 Hardware

10.2.2.1 Sensors

Load Alleviation - There is a variety of sensors which can be

used for load alleviation. The strain gage type sense load

variations directly; their sensitivity and frequency response

are quite adequate. The reliability record of this type of

sensors in an aircraft environment, however, is not well

established and even with utmost care in installation, the

sensor is one order of magnitude less reliable than the rest

of the system. At present Boeing-Vertol uses a strain gage

type of sensor in its cruise guide indicator to monitor rotor

loads. The angle of attack/yaw sensor (A(q) sensor) type

senses load variations indirectly by sensing the resultant

effect of loads on aircraft motions. The most commonly used

type is the differentia⁒ pitot-static pressure type used as a
sideslip sensor on the CH-46 and CH-47 production aircraft.
This device is very reliable and

has adequate sensitivity and frequency response. Its range

of operation, however, is above 40 knots airspeed, thus making

it unusable at hover and low speed operation. Another device

developed recently and not yet in production is a differential

airspeed sensor based on the generation of vortices by a strut

in the airstream at a rate which is proportional to the airspeed

past the strut. This is a very sensitive device and operates

at airspeeds down to 1 knot. Its frequency response is quite

adequate. The device is being tested by the Flight Dynamics

Laboratory and should have an established reliability record

in the time frame of interest for the stowed rotor aircraft.

Mode Suppression - The most practical and reliable type of
sensor available for mode sensing is an accelerometer. Units
are now available off-the-shelf with the required sensitivity
and high frequency response. Highly selective filtering must
be accomplished in order to discriminate mode frequencies from
the normal aircraft vertical accelerations due to gusts and
maneuvers. An alternate type of sensor is a sensitive strain
gage located on the wing skin. Filtering of the signal output
is again necessary to discriminate mode frequencies from
normal wing loads due to gusts and maneuvers.

10.2.2.2 Signal Processing

Electrical signal processing does not present any major
problem. Use of integrated circuits and micrologic permits
such functions as signal amplification, filtering, logic
switching and redundancy management to be accomplished without
the need for cumbersome and complex circuits. Multiple filters
of the Tchebyscheff or Butterworth types can be used for highly
selective filtering in the mode suppression feedback. Micro-
electronic components available now are highly reliable and
relatively inexpensive.

10.2.2.3 Actuators

The actuators required for SAS are high power, fast acting
differential actuators. The requirement can best be met by
electrohydraulic actuators. Two types have been considered
and found to be adequate. One is of the position summing

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type (total SAS piston movement and authority is the sum
of the piston movement and authority of the individual
actuator units); the other is of the force summing type
(total SAS force is the sum of the forces of each unit).

10.3 GOVERNOR SYSTEMS

Rotor speed governing is achieved by controlling the collective
blade pitch angle only; no engine power control is involved in
speed governing. The governor actuator is inserted differen-
tially into the collective linkage of the primary controls;
thus, a single actuator controls blade pitch on both rotors.
Integral control is used in order to achieve the objective
of a steady state governing accuracy of 0.3% of rotor speed.
The rotor speed demand signal is programmed as a function
of nacelle tilt angle. This demand signal is compared to
actual rotor speed signal and the difference is used to drive
the governor actuator. Provision can be made in the governor
to accept control quickening through a washed-out signal
from the longitudinal control stick transducer. The functional
block diagrams of three separate possible govern system config-
urations are shown in Figures 10.1, 10.2 and 10.3 and described
below. Each of the configurations is redundant in order to
comply with the failure safety requirements.

10.3.1 Dual Electromechanical Governor System

Figure 10.2 shows a dual electromechanical system with one
channel active and one in synchronized standby. Switching
from the active to the standby channel occurs automatically

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upon sensing of a failure in the active system. The logic
to sense a failure requires a third rotor speed sensor and
third feedback to determine by majority voting which of the
two channels has failed. Provision is made for manual oper-
ation of the governor after failure of both active and standby
channels and their subsequent shutdown. Manual operation is
possible only if the two failures do not involve either motor
or clutch in both channels. In manual operation, the rotor
blade pitch is positioned through a beep switch to the angle
providing the desired rotor speed. This mode of operation
obviously requires a substantial increase in pilot workload
and does not provide either accurate or fast rotor speed
control.

The system is thus fail functional at first failure; however,
safety of manual operation after second failure is not obvious,
in light of the required additional pilot workload to operate
the governor.

10.3.2 Triple Electrohydraulic Governor System with Model
Channel

Figure 10.3 shows a system consisting of three active channels
and a model channel. Each channel controls one section of
a triple electrohydraulic force summing actuator. Each
actuator section has a failure monitor/pressure feedback
valve which provides pressure equalization among the three
sections in normal operation, and if equalization fails to
occur, provides the failure signal to the channel logic.

The three active channels by themselves provide fail

functional capability at first failure and the possibility

of hardover at second failure. In order to provide a fail

functional capability at second failure, a model electrical

channel is added which is used only to detect a second

failure and to immediately shut down the failed channel. In

order to achieve the dual fail functional capability, three

independent hydraulic supplies are required, and an electrical

supply which has a dual fail functional capability. The

failure detection logic associated with the model channel is

relatively much more complex than the logic required to

detect first failure.

10.3.3 Quadruple Electromechanical Governor System

Figure 10.4 shows a system consisting of four active channels.

The actuator is quadruple electromechanical; each channel is

velocity summed through mechanical differentials in the

actuator. Failure in one channel is detected as a motor

runaway condition; if the channel fails dead, there is no need

to detect the failure and remove the channel from operation

in flight. A fail functional capability is achieved at both

first and second failure because any runaway of one motor

due to channel failure is compensated by the remaining

three or two motors until the failed channel is automatically

removed from operation. A quadruple system instead of a

triple system is chosen because the simplicity of the failure

detection logic in a quadruple system far outweighs the

additional complexity of a fourth motor in the electromechanical actuator. This is not the case in the electrohydraulic actuator system of Figure 10.2 where addition of a fourth section would make the actuator too costly, too heavy and too complex. No hydraulic power supply is needed for this system. The electrical supply must have a dual fail functional capability in order to achieve dual fail functional capability for the entire system.

10.4   MODIFICATIONS OF FLIGHT CONTROLS

10.4.1   Tilt Rotor Modified Flight Control System

A minimal change approach to modifying the proposed Model 222 Flight Control System to include full load alleviation/mode suppression and lift augmentation is shown on the flight controls schematic of Figure 10.5. Four dual extensible link actuators are added as shown. These are essentially the same units now used for roll and yaw SAS for A_1 (lateral) cyclic feedback actuators, and their function is to provide the capability of differential yaw control.

10.5   HARDWARE MODIFICATIONS

Assuming that the rate damping capability is present in the

aircraft operating through differential actuators in the

longitudinal, lateral, and directional axes, addition of the

load alleviation, lift gust alleviation and modal suppression

functions entails the following hardware changes.

10.5.1   Load Alleviation

Two dual extensible link actuators must be added in each of

the two nacelles for complete differential cyclic control.

The SAS electronics must have the added circuits to process

the load alleviation signal from the sensors. Dualized

load alleviation sensors comprising the angle of attack sensors

and the sideslip sensors must also be added.

10.5.2   Lift Gust Alleviation

In order to control the spoilers, the flaps and the elevator

individually, three sets of dual extensible links must be

added in the control linkages located so as to permit collec-

tive operation of the flaps and the spoilers. The required

circuitry for signal processing must be added in the dual SAS

electronic boxes. Dual A(g) sensors or dual accelerometers

must be mounted to sense lift gusts.

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10.5.3   Modal Suppression

Modal suppression can be added without addition of any actuators, since the actuators for mode suppression are the same as those used for load alleviation. The SAS electronic boxes must be modified to hold the necessary circuits to process the mode suppression signals. The sensors for mode suppression are dual wing tip accelerometers.

In summary, the addition of the above functions requires doubling of the complexity of the SAS electronics over that of the standard rate damping SAS, addition of four dual extensible link actuators and the mounting of dualized A(q) sensors, dualized accelerometers, dualized angle of attack sensors and dualized sideslip sensors.
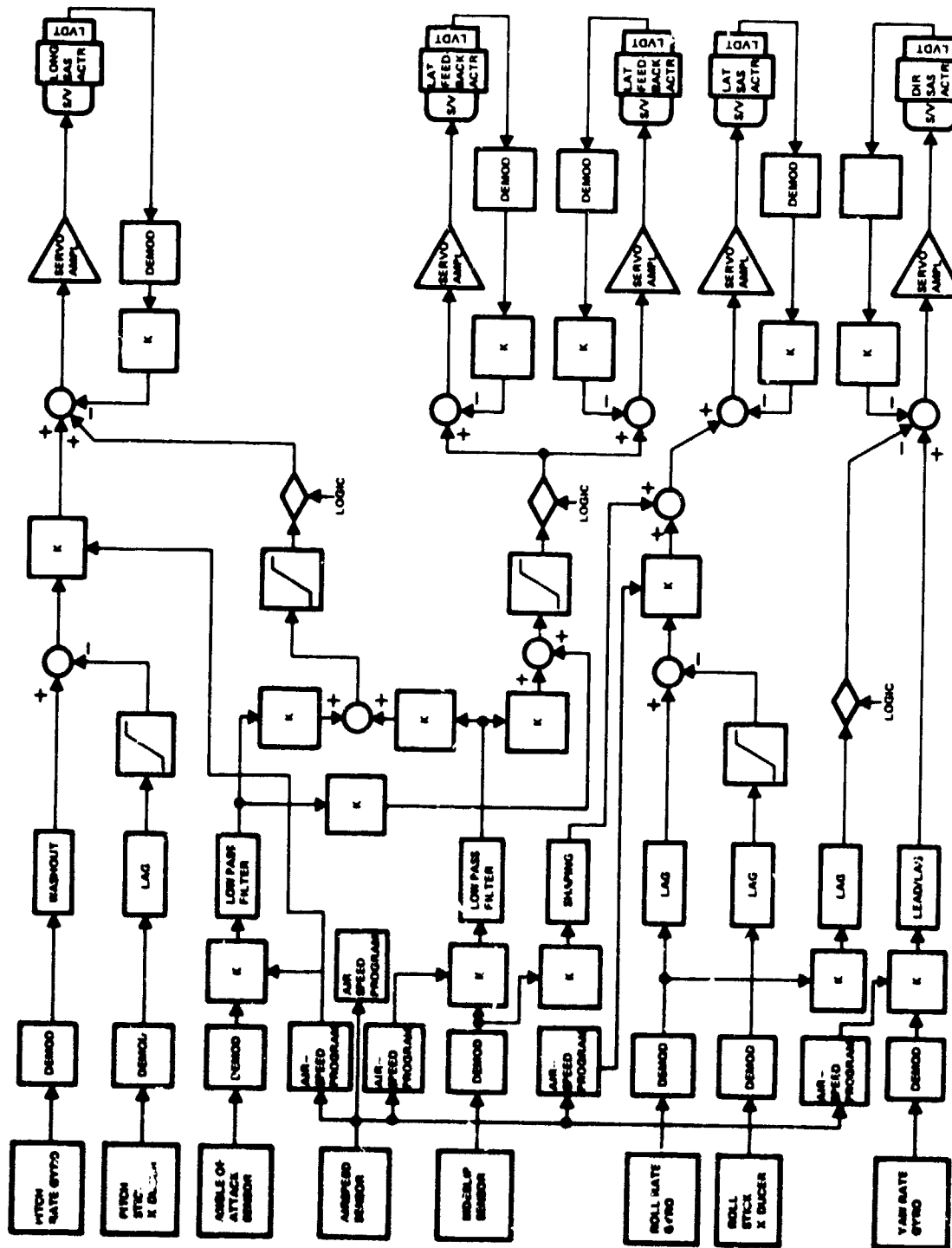
Figure 10.1 Stability Augmentation System – Functional Block
Diagram – One Channel
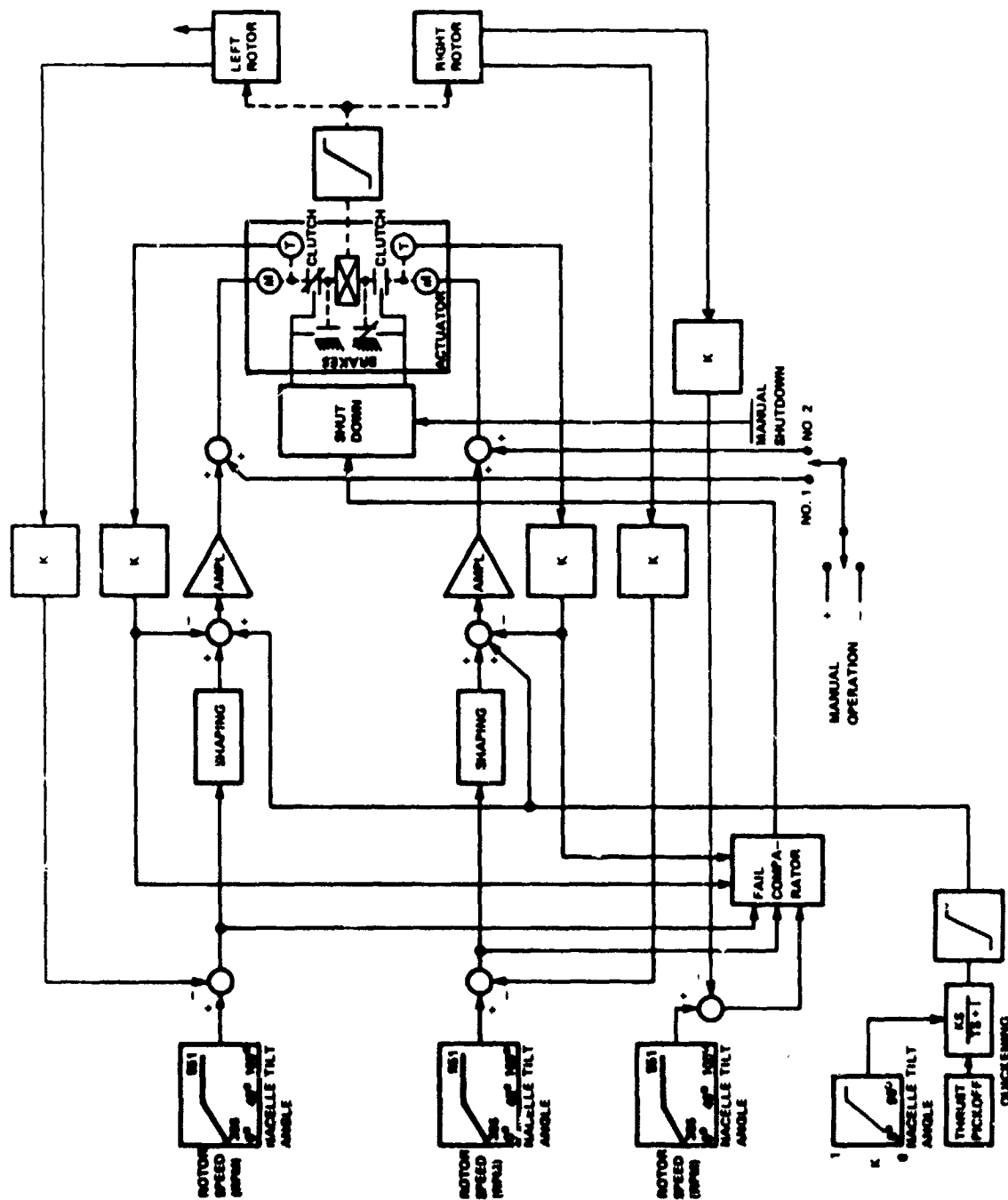
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Figure 10.2 Governor - Dual Electro-Mechanical System

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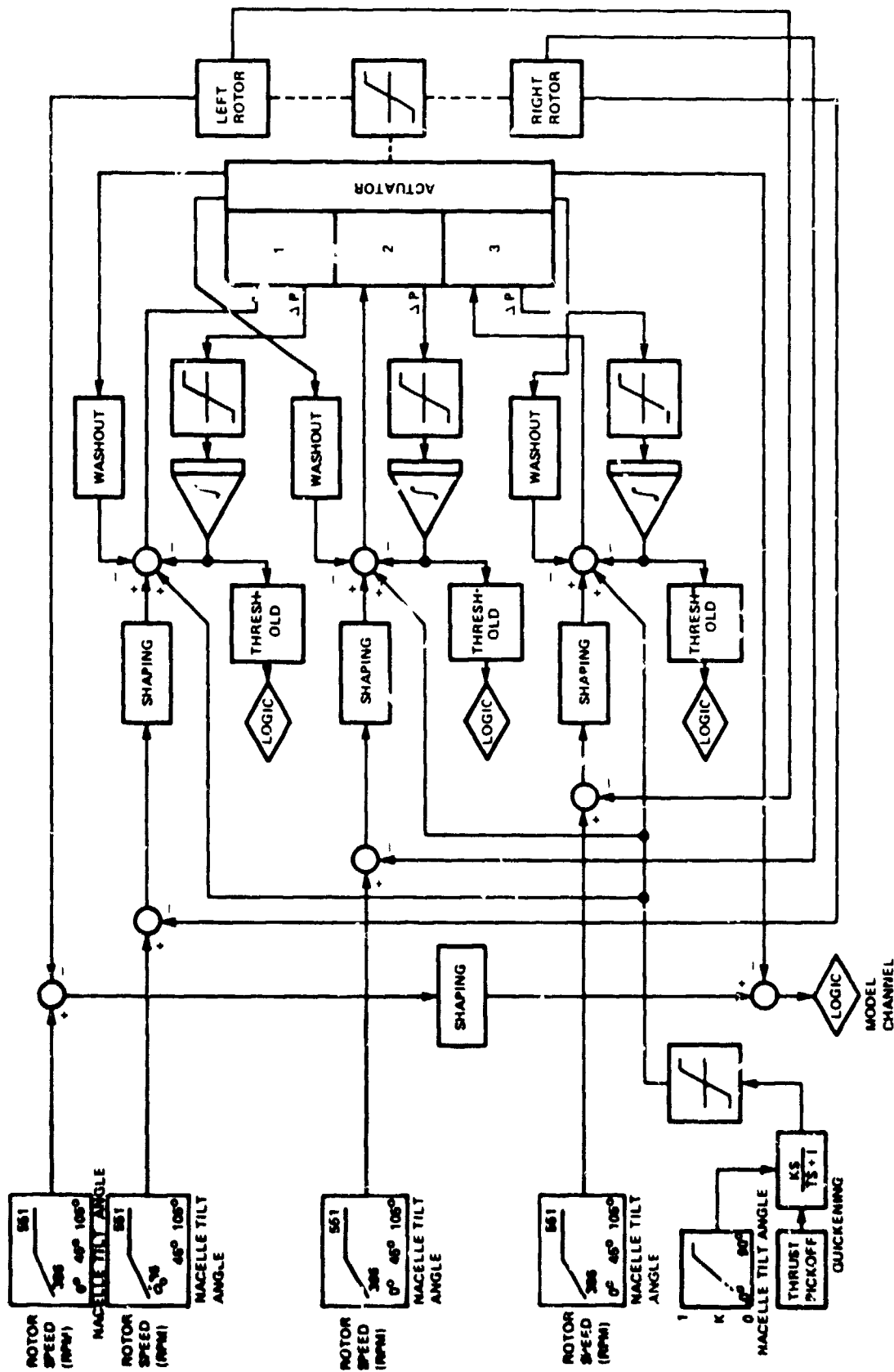
Figure 10.3   Governor - Triple Electro-Hydraulic
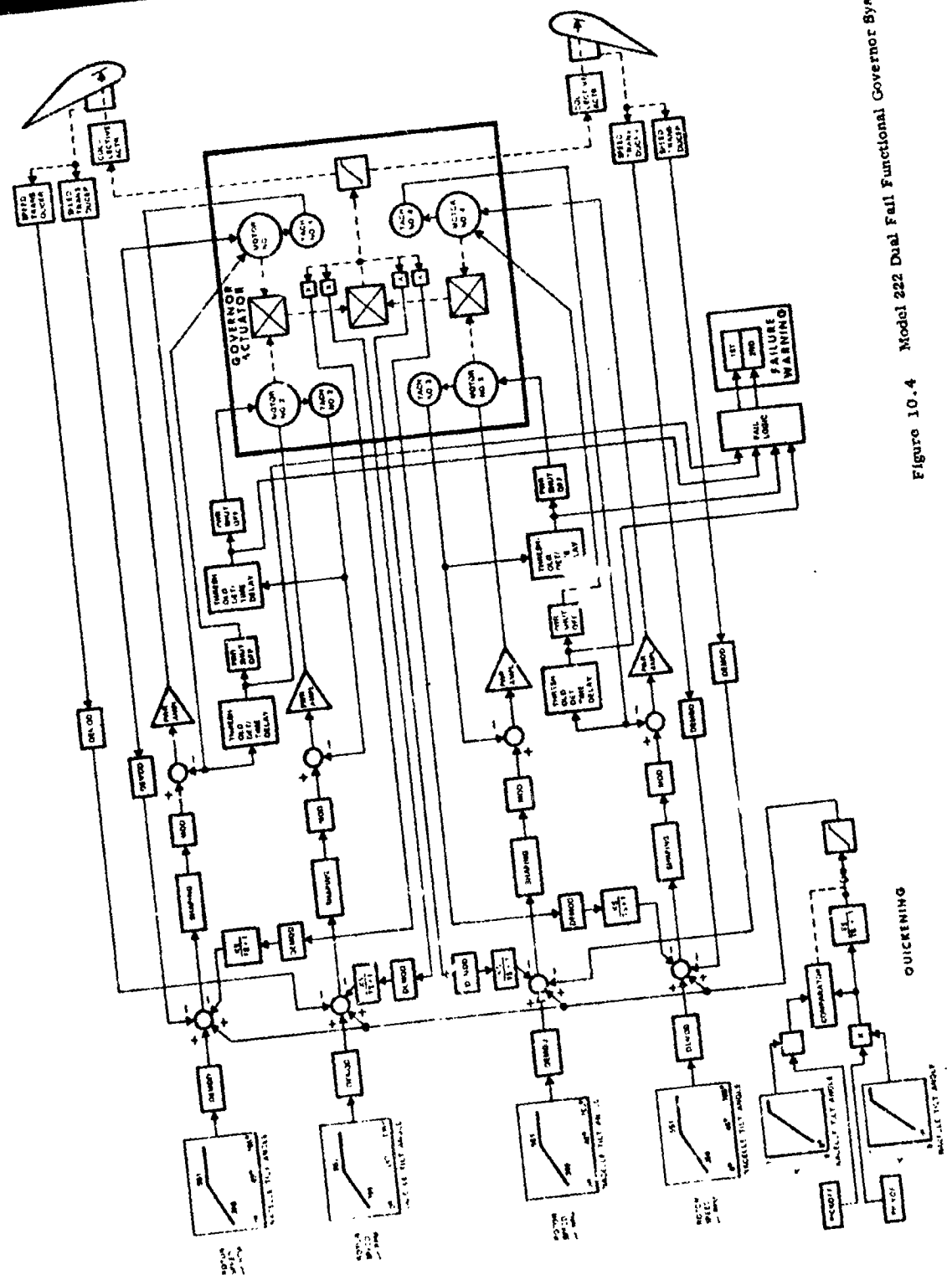
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Figure 10.4   Model 222 Dual Fail Functional Governor System

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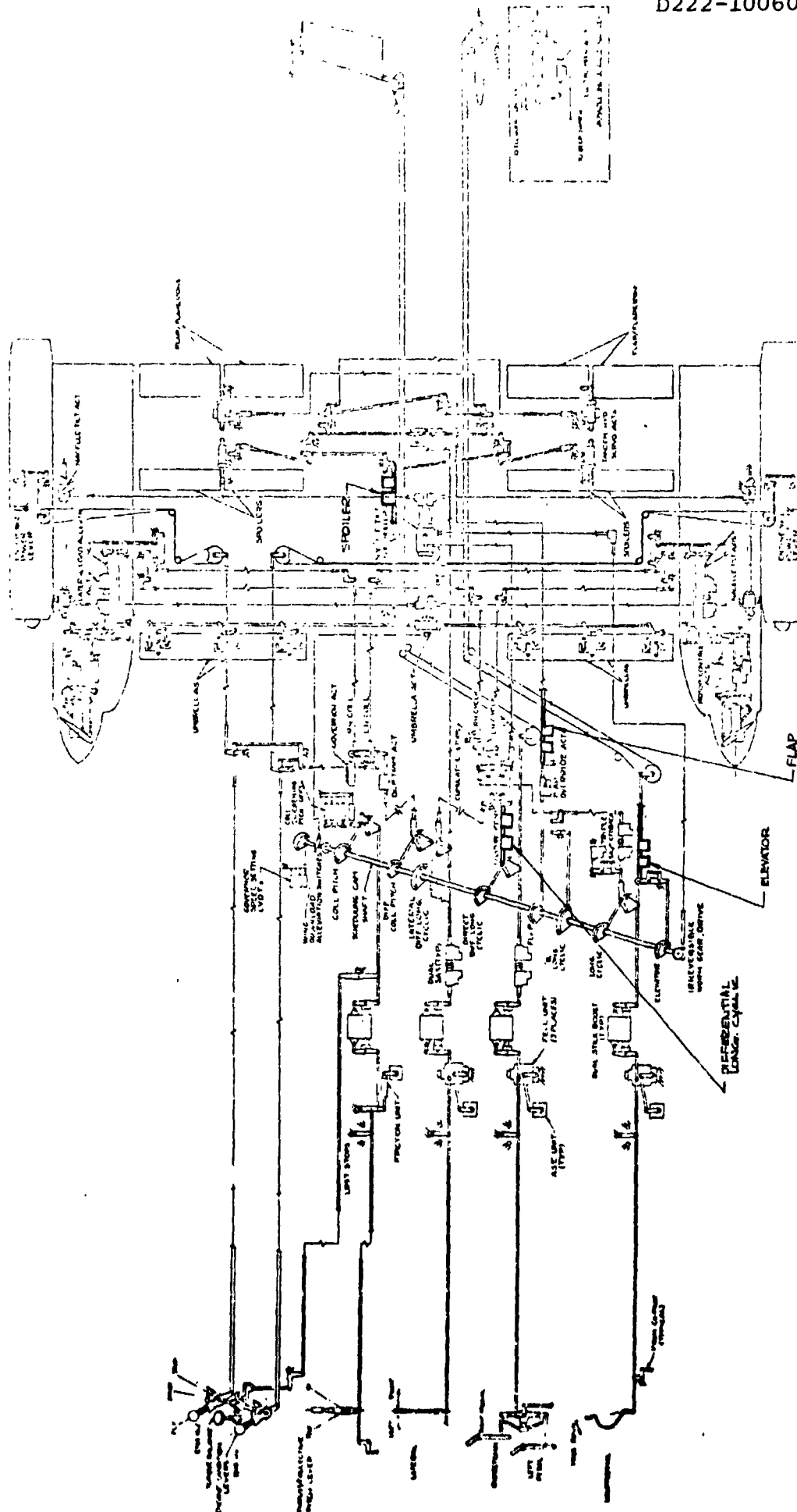
Figure 20.5   Flight Controls Schematic

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11.0 RECOMMENDED PROGRAMS

The programs of work recommended below fall under three major
headings. These are:

    (a) Analytical studies

    (b) Wind tunnel tests

    (c) Full scale flight test implementation

Discussion of (c) is divided into programs for which provi-
sion has been made in the Boeing proposal for a tilt rotor
research vehicle, and additional programs which would entail
modifications to the control system or additional sensing and
avionic components.

11.1 RECOMMENDED ANALYTICAL STUDIES

A group of recommended analytical studies are set out in
Table 11.1. These include investigations directed toward
more efficient use of swashplate feedback for load alleviation
and modal suppression, the combination of different types of
feedback control, and first cut feasibility studies
on control configured vehicles and systems where reliance is
placed on feedback to maintain aeroelastic stability.

In the course of the current contract, aspects of the load
alleviation systems and modal suppression systems emerged
which merit additional study. These include the resolution

TABLE 11.1 – RECOMMENDED ANALYTICAL STUDIES

System	Objectives	New Aspects	Possible Advantages	Methodology Requirements
1) Free flight blade load alleviation systems and stability augmentation	Control of blade loads in transient maneuvers and gusts	Evaluate use of transfer functions to control azimuth as function of frequency. Also phased A_1 and B_1 feedback.	More positive achievement of objectives, and removal of conflict between static and dynamic cases	Minor modifications to existing capability (C-48)
2) Aeroelastic stability and modal suppression	Reduction of modal response	Explore use of phased A_1 and B_1 cyclic and/or tip vanes or flaps for damping	More effective use of available cyclic. Reduction of side effects.	As above
3) Combined direct lift and swashplate feedback for blade loads alleviation	Explore whether more effective control of blade loads is possible using conventional control feedback along with cyclic feedback	Assessment of combined systems	Reduction of short period mode response and increased effectiveness of blade loads reduction systems	Modifications to existing capability to permit representation of direct lift and swashplate systems simultaneously
4) Control configured vehicle studies	Explore use of swashplate feedback to replace horizontal and vertical stabilizers	Partial or complete removal of empennage	Reduction in weight and drag	Minor changes to capability
5) Feedback for primary control of aeroelastic stability	Replace reliance on stiffness by feedback system	Parametric study of stiffness	Thinner and lighter wings	Minor changes

of azimuth requirements at different frequencies of loading.
In Section 3 it is shown that the feedback azimuth appropriate
to static conditions is significantly different to that
required for alleviation of a gust. A method of resolving
this conflict may be to control azimuth as a function of fre-
quency by the agency of different attenuating filters in the
A_1 and B_1 channels.

Another aspect which has emerged from the current study and
confirmed by tests is that a dynamic conception of the rotor
is required for adequate mathematical modelling. This is
particularly apparent in attempts to obtain pure normal force
and pitching moment feedback signals. It was found that under
oscillatory conditions no azimuth existed about which the
swashplate could be rotated to produce normal force without
an accompanying component of side force. It had been expected
from static experience that such an azimuth could be found,
but on reflection it is obvious that under dynamic conditions,
the swashplate will require a precession or wobble in order to
produce a force or moment vector which may be adjusted to give
normal force or pitching moment without an associated side
force and yawing moment. Such a precession is generated by
providing time phasing between the pitch and yaw cyclic signals

demanded by the control system. The advantages of such an arrangement are twofold:

(1) All the system authority (which is limited for several reasons) is directed to producing useful output, with a correspondingly more effective system

(2) Potentially adverse loadings are removed, i.e., the oscillatory side forces generated when only normal force is required

The studies reported in the current volume have not attempted to explore the potential of these concepts. It is proposed that they be the subject of a separate investigation which would include the following topics:

(a) Phase and relative A_1, B_1 gain optimization of a blade load alleviation system over a practical range of static and transient gust and maneuver conditions

(b) Phase and relative A_1, B_1 gain optimization of a modal suppression system under the same conditions

(c) Integration of both systems with a governing and direct lift feedback system and reevaluation of optimal settings

It is also proposed that this may be confirmed by wind tunnel test demonstration as outlined in Paragraph 11.2. An initial

and fairly inexpensive demonstration of the effectiveness of

phased swashplate feedback is proposed using a reactivated

1/9.26 scale model in the Princeton tunnel, and it is hoped

that by the time further model work using the 1/4.622 scale

model and full scale flight testing, the principles of phased

swashplate feedback would be well established, and be demon-

strated further.

In the work to date, feedback has not been considered a primary

means of ensuring aeroelastic stability, but only as a modal

suppression or damping augmentation system. It is now pro-

posed to explore the potential benefits in weight and per-

formance that would accrue from the use of feedback to relax

wing stiffness requirements and permit the use of thinner and

lighter wing structures.

An additional concept which merits analytical investigation in

the near term is the control configured tilt rotor vehicle.

It is considered that a stage of development and validation of

analytical methodology relating to tilt rotor feedback control

systems has been reached, where studies of advanced control

configured vehicles may be profitably undertaken. The controls

configured tilt rotor system would use rotor forces and moments

to provide the stability and control currently envisioned

to come from the empennage and empennage controls. Thus,

longitudinal and directional stability and control would be
provided by rotor forces and moments only. Lateral directional
control and stability would be provided by differential rotor
forces and moments including differential collective and ail-
eron and spoiler motions. For example, the yaw damping and
stiffness effectiveness of a vertical stabilizer would be
replaced by differential collective signals proportional to
rate of sideslip and angle of sideslip respectively. The
dynamics of the rotor are an integral component of any such
system and it is believed that a state-of-the-art has now
been demonstrated where meaningful analytical studies may be
undertaken. These studies would determine the control ranges
required, the feedback system characteristics in terms of gains
and authorities and would investigate the effect of the systems
on the rotor blade stresses. Required nacelle tilt rates and
the effect of these on the nacelle tilting system would be
investigated. These would logically preceed attempts at
tunnel demonstrations and full scale design.

11.2 PROPOSED WIND TUNNEL PROGRAMS

A number of wind tunnel programs are proposed in Table 11.2.
These envision tests on the 1/4.622 scale model of the M222
aircraft and further testing of the 1/9.622 scale model dis-
cussed earlier to investigate new system concepts.

TABLE 11.2   PROPOSED WIND TUNNEL PROGRAMS

	Test	Model	Tunnel	Test Objectives	Special Requirements
1.	Blade Load Alleviation Study	1/4.622 Scale	BV or suitable alternative (with gust generator)	Develop and demonstrate load alleviation system. Evaluate transfer function control of azimuth as function of frequency. Also phased A_1 and B_1 feedback.	Monkey pole suspension
2.	Rotor Governing and Blade Load Alleviation	1/4.622 Scale	BV or suitable alternative (with gust generator)	Demonstrate satisfactory operation	None
3.	Modal Suppression System	1/4.622 Scale	BV or suitable alternative (with gust generator)	Design and demonstrate modal suppression system using rate and deflection signals and phased A_1 and B_1 feedback	Test for compatibility with lift control system
4.	Direct Lift Control	1/4.622 Scale	BV or suitable alternative (with gust generator)	Acquire spoiler and flap oscillatory lift, pitching moment and downwash data design lift control system. Demonstrate.	Modifications to current design balance and monkey pole testing
5.	Phased Swashplate Feedback and Azimuth Control As Function of Frequency	1/9.622 Scale	Princeton Tunnel	Explore use differential A_1 and B_1 transfer function and wobbling swashplate concept to reduce blade loads and suppress modal response	Existing model with minor mods to electronics

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The tests proposed for the 1/4.622 scale model are in two groups. The first three use currently defined hardware and may be performed with little or no modification to the existing model. The fourth which is intended to demonstrate a direct lift system would require control system modification to permit symmetric actuation of flaps, spoilers and elevators. Reactivation of the 1/9.622 scale model involves minimal hardware modifications since only the electronic processing of feedback signals would need changing.

The 1/9.622 scale model test would investigate the use of the concepts of signal shaping discussed in Section 11.1. The 1/4.622 scale tests would involve demonstration of currently defined systems and also the new concepts of signal shaping discussed and demonstrated in the 1/9.622 scale test. Programs 1, 2 and 3 might profitably be merged into one test while number 4, the direct lift program, might entail a second installment. The first section (Tests 1, 2, 3) probably represents a 2-3 week period of tunnel occupancy and test 4 is probably a similar time span since the swashplate systems settings would be reoptimized during this test.

The 1/9.622 test is a research oriented activity and will, therefore, also require approximately a 2-3 week tunnel occupancy.

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11.3 PROGRAMS PLANNED AND ADDITIONAL RECOMMENDED PROGRAMS: FULL SCALE FLIGHT TEST

Recommended full scale flight test programs associated with feedback controls are listed in Table 11.3.

11.3.1 BLADE LOAD ALLEVIATION SYSTEM AND STABILITY AUGMENTATION ON THE NASA TILT ROTOR RESEARCH VEHICLE

The NASA Tilt Rotor Research Aircraft is configured to include a load alleviation swashplate feedback system. The system as proposed by Boeing Vertol senses angle of attack and sideslip and processes these signals to give swashplate A_1 and B_1 cyclic responses; the objective is to reduce the blade loads induced by trim changes, gusts and maneuvers and concurrently improve aircraft dynamic stability. The system will be flight tested and optimized. The flight test program of development and testing will be preceded by scaled model testing to demonstrate the system. The model testing is discussed in Section 11.2.

11.3.2 DIRECT LIFT FLIGHT PATH CONTROL AND GUST ALLEVIATION

The rotor swashplate systems discussed are of limited force level and have significant time lags due to rotor response. The alleviation of "g" response to high frequency turbulence requires a more powerful and faster acting system which provides a direct feedback on aircraft lift. It is proposed that the flap and spoiler controls be modified to accept symmetric

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TABLE 11.3   RECOMMENDED PROGRAMS ON FULL SCALE AIRCRAFT WITH SUPPORTING MODEL TEST PROGRAMS*

	SYSTEM DESIGNATION	OBJECTIVE	CURRENTLY PLANNED FOR NASA RESEARCH VEHICLE?	GROUND TESTING	FLIGHT TESTING	SUPPORTING MODEL PROGRAMS	AIRCRAFT MODIFICATIONS AND SPEC. HARDWARE
1.	LOAD ALLEV. FEEDBACK SYSTEM	REDUCE BLADE LOADS/IMPROVE A/C STABILITY CHARACTERISTICS	YES	FUNCTIONAL CHECKS. FREQUENCY RESPONSE STABILITY CHECKS	DEVELOP AND DEMONSTRATE SYSTEM	TUNNEL DEMONSTRATION ON SCALED MODEL	HARDWARE INCLUDED IN M222 TILT ROTOR RESEARCH AIRCRAFT PROPOSAL
2.	DIRECT LIFT GUST ALLEVIATION	ATTENUATE g LEVELS IN TURBULENCE. IMPROVE APPROACH HEIGHT CONTROL	NO	FUNCTIONAL CHECKS AND FREQUENCY RESPONSE STABILITY CHECKS FULL SCALE DATA ACQUISITION IN 40X80 TUNNEL	DEVELOP AND DEMONSTRATE SYSTEM	1.ACQUIRE TUNNEL DATA ON LIFT VARIATION RESPONSE OF FLAPS, SPOILERS AND ELEVATORS. 2.SCALED MODEL DEMONSTRATION 3.FULL SCALE CONFIRMATION OF FLAP/SPOILER/ELEVATOR DATA.	MODIFIED FLAP AND SPOILER SYSTEM. SENSORS AND AVIONICS.
3.	MODAL SUPPRESSION SYSTEM USING SWASHPLATE FEEDBACK	AUGMENT DAMPING IN STRUCTURAL MODES	NO	FUNCTIONAL & FREQ. RESPONSE FULL SCALE DEMONS. IN 40 X 80	DEVELOP AND DEMONSTRATE IN FLIGHT	1.ADDT'L. TESTING WITH 1/9 SCALE MODEL, PRINCETON. 2.1/4 SCALE MODEL DEMONSTRATION	SENSORS AND AVIONICS. EXISTING ACTUATORS PROBABLY O.K.

* SEE TABLE 11.2 FOR DETAILS OF SCALED MODEL PROGRAMS.

commands from an angle of attack sensor. The system will also command suitable amounts of phased elevator travel to reduce pitching response caused by the gust and flap and spoiler operation.

The effectiveness of such a system would be examined on a scaled model prior to a full scale commitment and as indicated in Table 11.3 model or full scale tests to acquire empirical data on oscillating flap and spoiler lift and downwash is probably advisable prior to final definition of the system.

11.3.3 MODAL SUPPRESSION SYSTEMS BASED ON ROTOR/SWASHPLATE FEEDBACK

A recommended program which is a ready adaptation from the load alleviation system is the modal suppression or aeroelastic stability augmentation system. The system has a substantial number of hardware items in common with the load alleviation/ stability augmentation system discussed above. Common swash-plate actuators may be used with different sensors and signal shaping. The ability of such a system to reduce wing response in turbulent air would be explored and damping measurements made. Such a system has been demonstrated in the 40 X 80 tunnel at NASA Ames, and very substantial increases in wing damping shown to be available. The system envisioned here wou'd be more advanced in certain respects than those tested to date.

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The chief differences will be in the type of feedback signal used and in the phasing of the pitch and yaw cyclic commands. It is believed that a more efficient use of the available control authority may be brought about by the use of phased A_1, B_1 or wobbling swashplate control. The modal suppression system by reducing the wing response will attenuate the associated cabin accelerations due to turbulent air, and will thus supplement the direct lift control system, which is highly effective in counteracting gust force on the overall aircraft, but is not necessarily effective in reducing modal responsiveness.

11.4   PLANNING COSTS AND SCHEDULES

Planning costs and schedules are presented in this section for a program which will evaluate the following systems:

- . Load alleviation feedback
- . Direc: lift gust alleviation
- . Modal suppression systems using swashplate feedback

The planned program will include flight tests and will be supported by analytical studies and scaled model wind tunnel tests.

11.4.1   ANALYTICAL STUDIES

Studies will be performed to support the planned program including pretest predictions and safety analyses.   In addition, investigations will be made of control configured vehicle surface studies, and feedback for primary control of aeroelastic stability.   The planned program does not include the implementation of the results of the additional studies.

11.4.2   DESIGN AND HARDWARE MODIFICATION AND INSTALLATION

The tilt rotor aircraft's flight control system will be modified to incorporate the planned systems (including sensors, actuators and SAS control box circuits).   Components will be bench checked and the modified and/or added components will be

installed on the aircraft at offsite Government facilities.

11.4.3 MODEL WIND TUNNEL TESTING

A contractor's tilt rotor model will be refurbished and modified to incorporate the planned systems modifications. Three test programs will then be conducted to determine the following characteristics:

- Blade load alleviation
- Rotor governing and blade load alleviation
- Modal suppression system
- Direct lift control
- Phase swashplate feedback

The tests will be conducted at the contractor's wind tunnel and at the Princeton facilities. The gust generator required for one phase of model testing has not been priced in this plan. It was assumed one would be available in the 1977 time frame. Tunnel occupancy will be for 280 hours and the total test period will be seven (7) weeks.

11.4.4 GROUND AIRCRAFT TESTS

The modified control system in the aircraft will be proof loaded. Functional and frequency response stability checks will also be made.

11.4.5 FLIGHT TESTS

The modified control systems will be demonstrated in a four
(4) week flight test program after completion of NASA tests
on the tilt rotor research aircraft in the 1978 time frame.
Testing will be performed by NASA with Boeing Vertol providing
supporting personnel. No 40' x 80' wind tunnel testing is
included in this estimate.

The planning costs presented are based on projected CY 1977
planning dollars which is intended to represent an average
for the period of performance:

	M.H.	COST $
ANALYTICAL STUDIES & PRETEST PREDICTIONS		
ENGINEERING	8,500	277,525
HARDWARE MODIFICATION		
DESIGN		
ENGINEERING	2,734	89,265
FABRICATION & INSTALLATION		
ENGINEERING LIAISON	560	18,284
DEVELOPMENTAL LABOR	1,644	38,075
PLAN. LIAISON, PROD. SUPPORT & TOOL SERVICES	253	5,976
QUALITY CONTROL	181	4,442
MATERIAL	——	13,780
	2,638	169,822

	M.H.	COST $
WIND TUNNEL TESTING		
ENGINEERING	12,178	397,612
DEVELOPMENTAL LABOR	7,595	175,900
PLAN, LIAISON, SERVICES	1,110	27,635
QUALITY CONTROL	835	. 20,490
MATERIAL		22,750
	21,778	644,387
GROUND TESTS		
ENGINEERING	640	20,896
FLIGHT TESTS	480	15,672
MANAGEMENT DATA REDUCTION & REPORTS	3,920	127,988
	5,040	164,556
TOTAL RECOMMENDED PROGRAM	40,690	1, 256,290

TASK C - TILT ROTOR FLIGHT CONTROL SYSTEM

D222-10060-3

FIGURE 11.1 SCHEDULES - RECOMMENDED PROGRAMS

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2.1   Alexander, H., Amos, A., Tarzanin, F., Taylor, R., V/STOL
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      Report AFFDL-TR-72-40, May 1972.

3.1   Fry, B., Feedback Control Tests on a Windmilling 2.81 Foot
      Diameter Soft-in-Plane Hingeless Tilt-Rotor in the Cruise
      Mode, Boeing Document D222-10047-1, November 1972.

3.2   Magee, J., Wind Tunnel Tests of a Full Scale Hingeless
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5.1   Arnold, J. Supersonic Transport Longitudinal LAMS SAS
      Conceptual Study, Boeing Document D3-7600-10, May 1969.

5.2   Arnold J. and Townsend, J., Supersonic Transport Lateral
      LAMS SAS Conceptual Study Results, Boeing Document D3-
      7600-11, May 1969.

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      Stability Analysis - C-48 Users Document, Boeing Document
      D210-10435-1, April 1972

5.4   Alexander, H. R., Morris, J. and Spittle, R., Pretest Cal-
      culation of Open Loop Frequency Response and Stability of
      High Rate and Low Rate Swashplate Feedback Systems of M222
      26' Rotor Test, Boeing Document D160-10019-1, October 1972.

7.1   Deckert, W. and Ferry, R., Limited Flight Evaluation of
      the XV-3, Air Force Report AFFTC-TR-60-4, May 1960.

7.2   McFarland, R., Wind, NAPS-80 NASA-Ames Program Specifica-
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8.1   VanWagensveld,D., et al; Wind Tunnel Tests of the Aerodynamics
      and Dynamics of Rotor Spin Up, Stopping and Folding on a
      Semi-Span Folding Tilt Rotor Model. Air Force Report
      AFFDL-TR-71-62, Volume VII, October 1971.

APPENDIX A - DESCRIPTION OF THE MATH MODEL

The math model used to evaluate the various feedback configurations was designed with emphasis on simplicity so that any affects of the feedback could be easily evaluated. At no point, however, was the model stripped of the necessary blade, structural, and rigid body modes so that it was not fully representative of a tilt rotor vehicle with a soft in-plane rotor. In cruise, aerodynamics were tailored to yield flying quality modes which were representative of this style aircraft. In hover the aerodynamics represented a case with all flaps down and all umbrellas fully open (i.e., no fuselage or wing aerodynamics). A normal evaluation case would contain twelve to fifteen degrees of freedom including blade elastic modes, wing vertical bending, wing chord bending, wing torsional bending, any of the six rigid body freedoms, and generalized feedback freedoms. A more complete description of the math model is included in the C-48 User's Document (D210-10435-1).

Physical characteristics of the math model are similar to the Boeing Model M222. Figure A.1, A.2, and A.3 show the rotor geometrical properties and the blade frequency properties from various RPM and collective pitch settings. Figure A.4 delineates the wing uncoupled modal frequencies.

FIGURE A.1 MODEL 222 — ROTOR BLADE PROPERTIES

WEIGHT DISTRIBUTION — LB/IN

x/R	$\Delta W/\Delta X$, LB/IN
1.0 (TIP)	1.422
.975	1.422
.975	.395
.95	.395
.90	.395
.85	.404
.80	.684
.75	.406
.70	.409
.60	.414
.50	.422
.45	.436
.30	.313
.25	.308
.20	.303
.15	.356
.10	.486
.072	.583
.072	6.4
0 (ROOT)	6.4

TWIST DISTRIBUTION — DEG

x/R	θ, DEG
1.0 (TIP)	-7.79
.975	-7.79
.9	-5.19
.8	-1.73
.75	0.
.70	1.73
.6	5.19
.5	8.66
.4	12.12
.35	13.85
.3	16.10
.25	18.75
.2	22.20
.15	27.85
.10	33.38
.072	33.38
0 (ROOT)	33.38

BLADE FLAP EI

x/R	$EI_\beta \times 10^{-6}$, LB.IN.^{2}
1.0	3.78
.9	3.86
.8	3.84
.7	4.21
.6	4.31
.5	4.92
.45	5.50
.3	7.54
.2	10.50
.15	13.82
.10	27.55
.072	49.16
.072	$1. \times 10^6$
0 (ROOT)	1×10^6

BLADE LAG EI

x/R	$EI_r \times 10^{-6}$, LB.IN.^{2}
1.0 (TIP)	535.
.9	535.
.8	520.
.7	510.
.6	510.
.5	520.
.45	540.
.3	41.57
.2	35.00
.15	36.76
.10	44.71
.072	50.98
.072	$1. \times 10^6$
0 (ROOT)	$1. \times 10^6$

Radius = 156 Inches
Chord (Constant) = 18.85 Inches

FIGURE A.2 · MODEL 222 ROTOR BLADE VARIATION OF 2ND BENDING NATURAL FREQUENCY WITH ROTOR SPEED AND COLLECTIVE PITCH

MODEL 222 TILT ROTOR RESEARCH AIRCRAFT
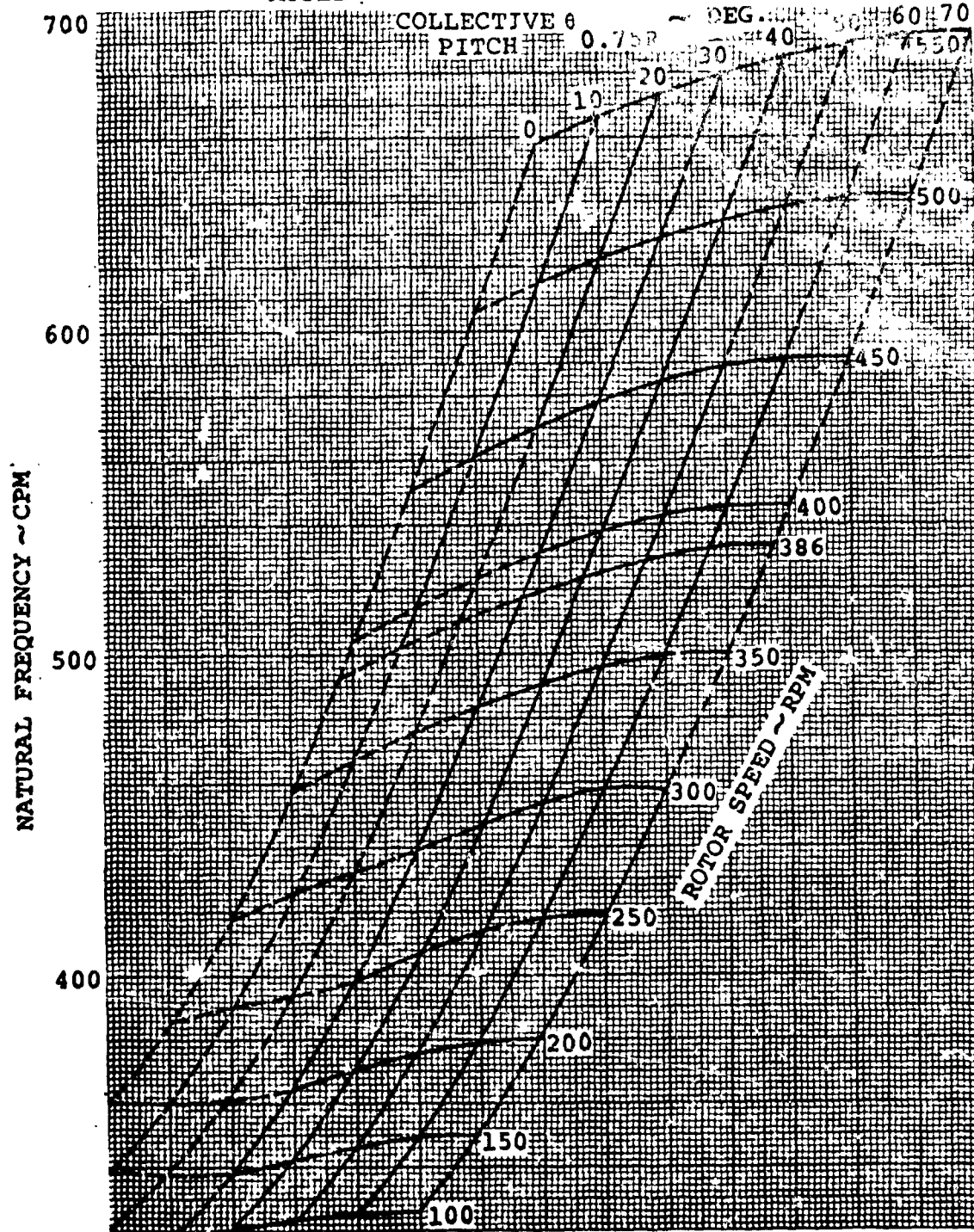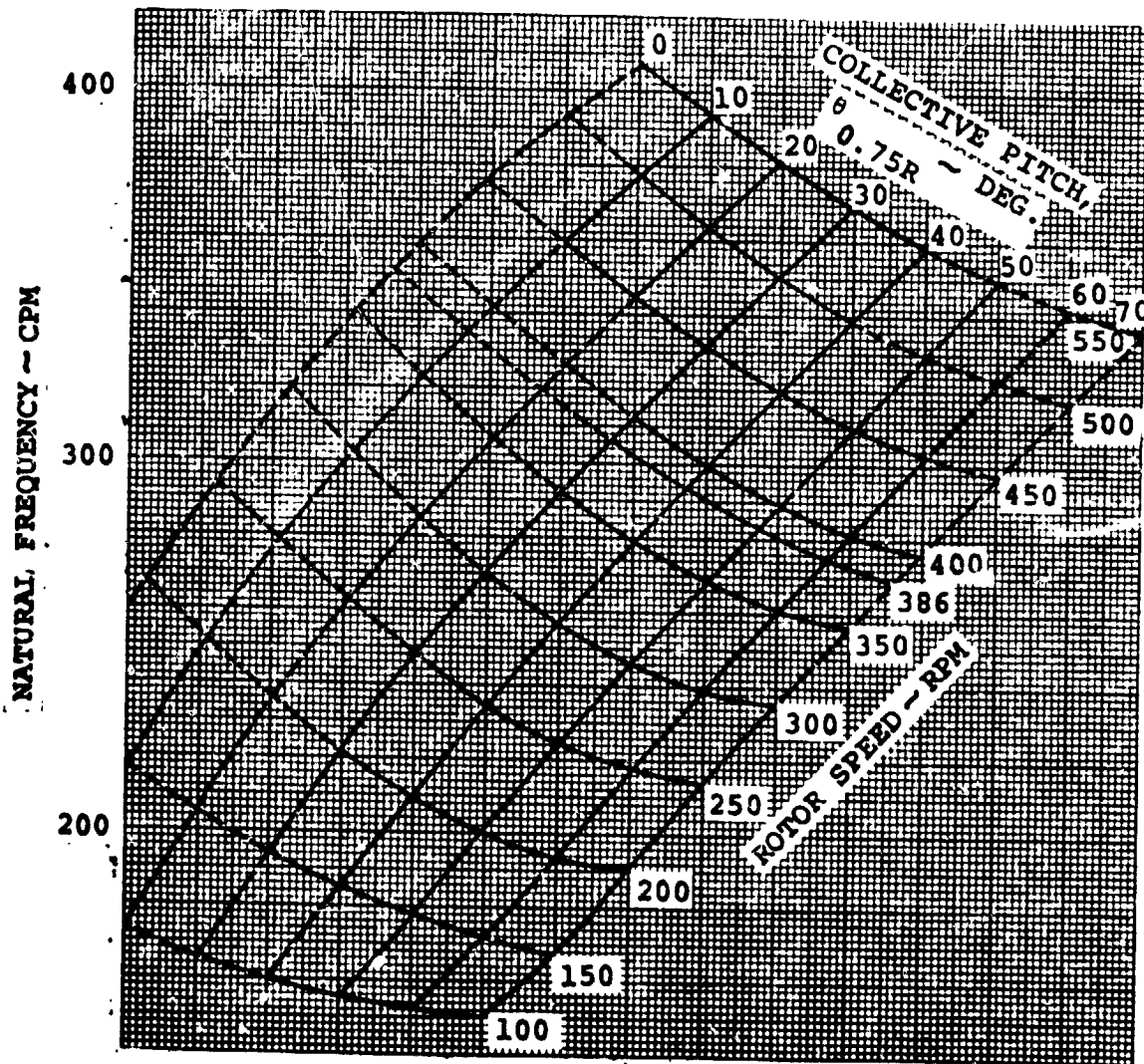

FIGURE A.3 - MODEL 222 ROTOR BLADE VARIATION OF 1ST BENDING NATURAL FREQUENCY WITH ROTOR SPEED AND COLLECTIVE PITCH

MODEL 222 TILT ROTOR RESEARCH AIRCRAFT

SYMMETRIC MODE	FREQUENCY
VERTICAL BENDING	3.6 CPS
CHORDWISE BENDING	5.4 CPS
TORSION	6.1 CPS

ANTISYMMETRIC MODE	FREQUENCY
VERTICAL BENDING	11.2 CPS
CHORDWISE BENDING	9.1 CPS
TORSION	5.7 CPS

FIGURE A.4   WING UNCOUPLED FREQUENCIES (BLADES OFF)
CRUISE CONFIGURATION

APPENDIX B - SINGLE GOVERNOR WITH TWO SENSORS BODE ANALYSIS

Bode analysis was made of the governor feedback loops for RPM feedback in various flight conditions to determine the maximum gains and need (if any) for filtering of high frequency signals. A criteria of 30-degrees phase margin or 6 dB gain margin (whichever is most stringent) was used. These margins are well recognized in the literature (for example, Saucedo and Shiring - Introduction to Continuous and Digital Control Systems) as limits which yield good transient and steady-state response.

The following definitions of gain and phase margin were applied.

- o the gain margin is the reciprocal of the gain at the frequency at which the phase angle reaches 180-degrees

- o the phase margin is the phase angle in degrees which the phase diagram must be shifted in order to obtain 180-degrees phase angle at unity gain.

All governor components in the block diagram of Figure have been represented except the sensor dynamics. Sensor frequency response is essentially "flat out" to 50 cps and spot checks of the Bode diagrams indicated that the sensor dynamics did not influence the maximum gain levels.

Levels of gain specified on the Bode plots correspond to values of G on the block diagram and have units of volts per

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volt. They are not to be confused with values of loop gain
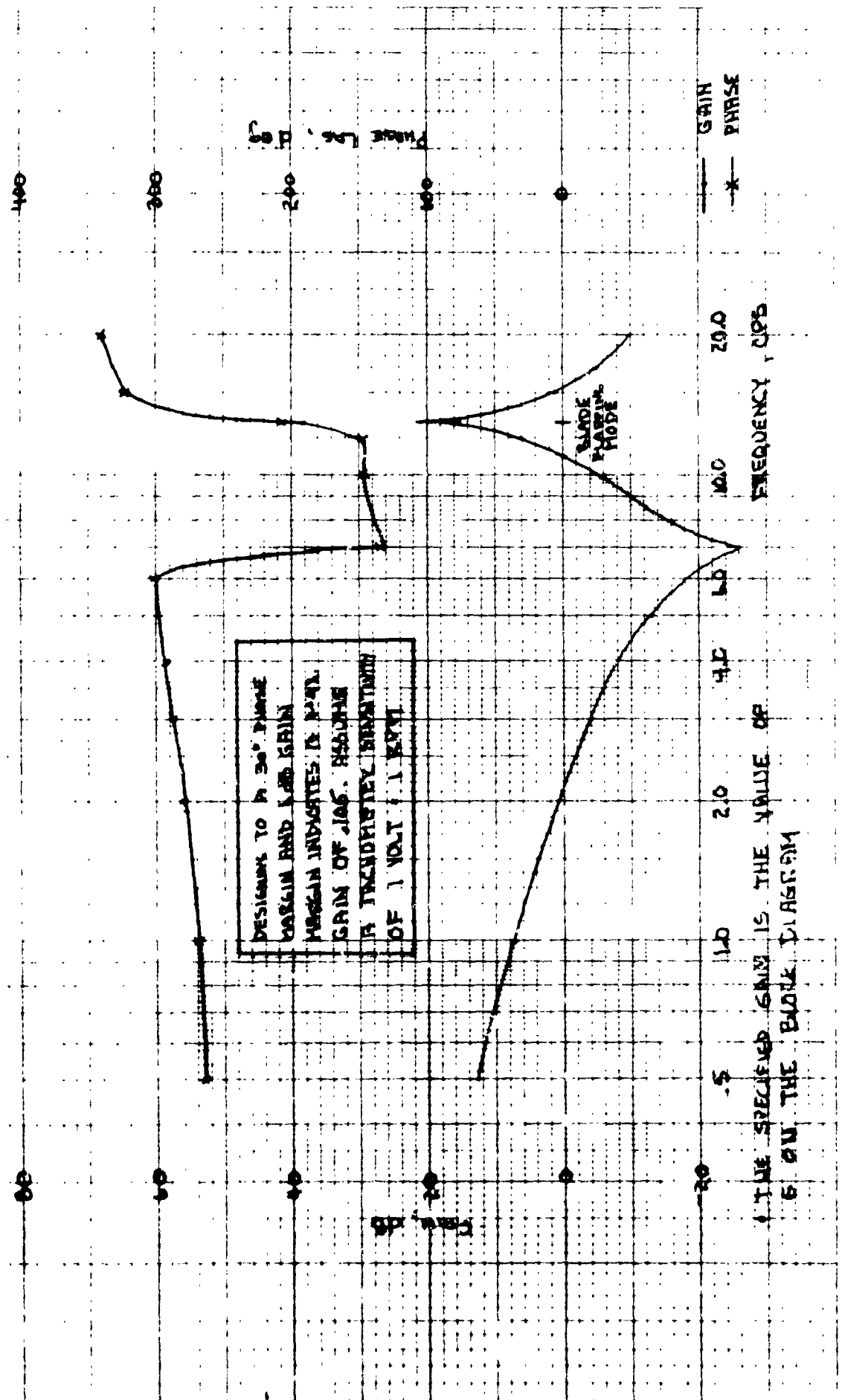
which have units of degrees per RPM.

The harmonic response was obtained using an option of the

C-48 Flying Qualities and Aeroelastic Stability Program. A

description of the program is contained in Boeing Document

D21C 10435-1.

REFERENCES

B.1 Saucedo, R., and Schiring, E., Introduction to Continuous and Digital Control Systems, The MacMillan Company, New York, 1970.

HOVER
551 RPM
NO X SHAFT

FIGURE 81 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

Phase Lag, deg

FREQUENCY, CPS

BLADE FLAPPING MODE

DESIGNED TO A 30° PHASE
MARGIN AND 6 db GAIN
MARGIN INDICATES A FULL
GAIN OF .106. RESPONSE
OR TACHOMETER ROTATION
OF 1 VOLT = 1 RPM

GAIN

PHASE

Gain, db

* THE SPECIFIED GAIN IS THE VALUE OF
G ON THE BLOCK DIAGRAM

FIGURE 8.2 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

FIGURE 8.15 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

HOVER CONFIGURATION
CROSS SHAFT PRESENT
20 FPS DESCENT
551 RPM

FIGURE 5.4 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

FIGURE B.5 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

HOVER CONFIGURATION
CROSS SHAFT PRESENT
20 FPS CLIMB
551 RPM

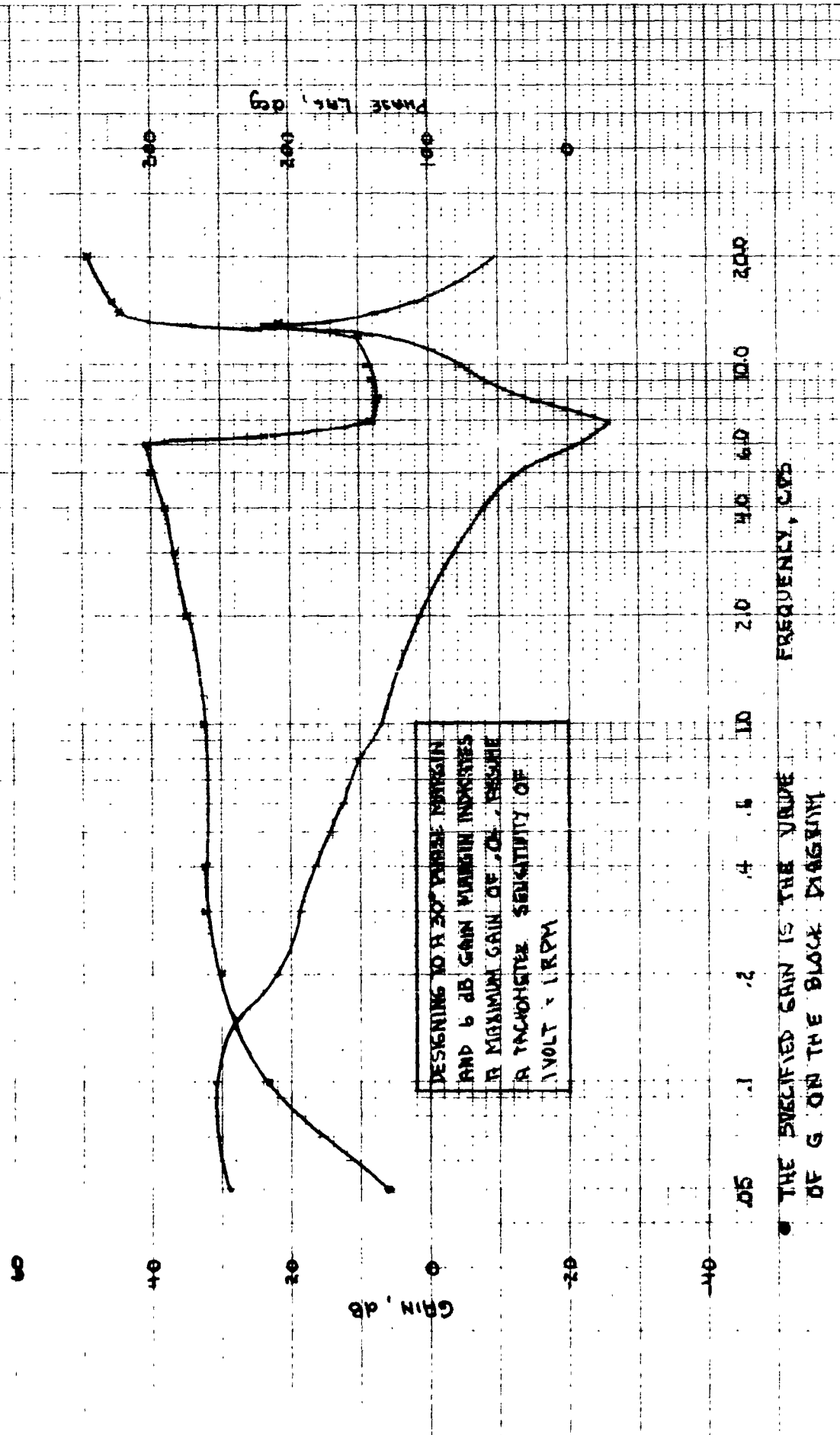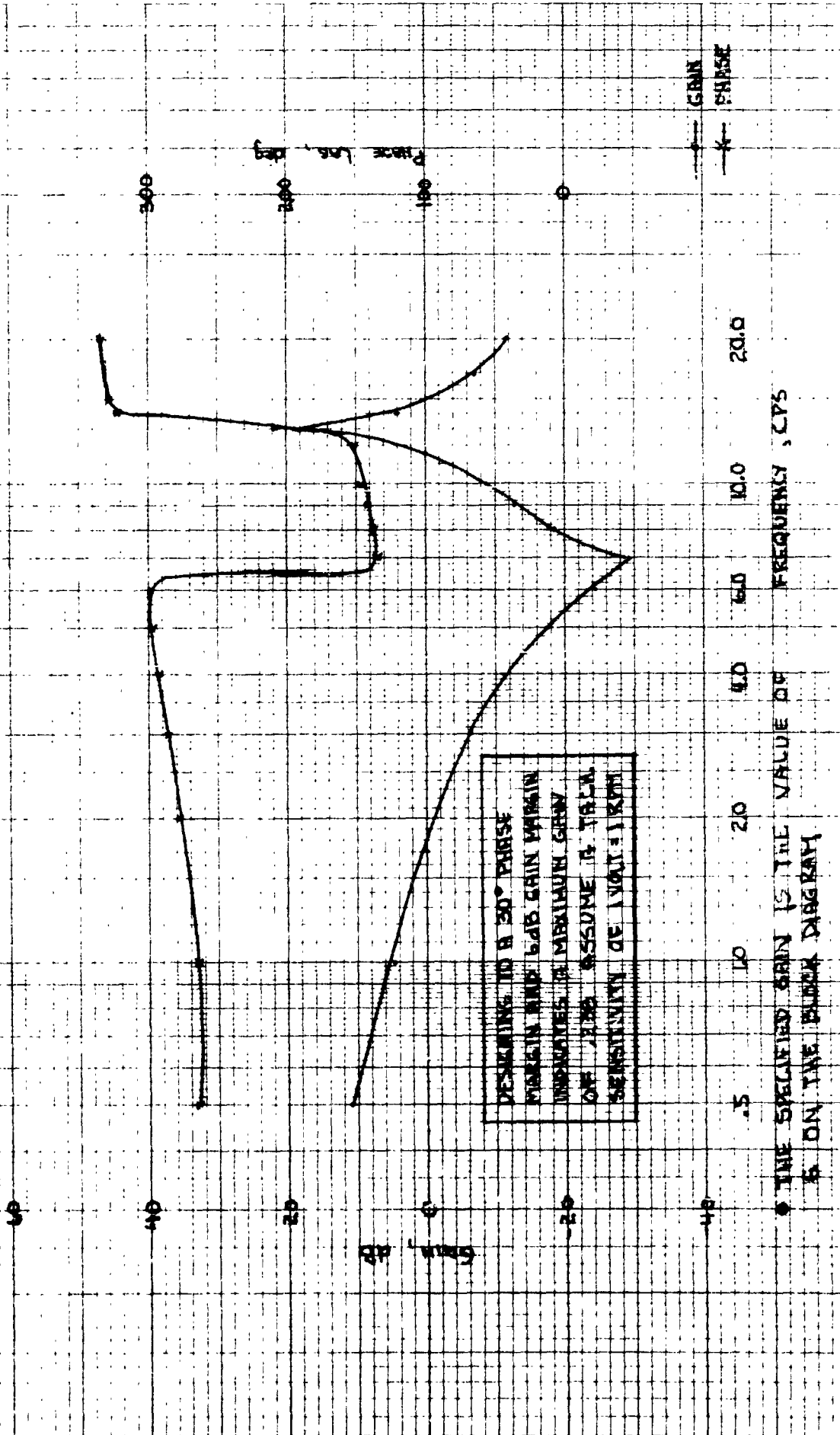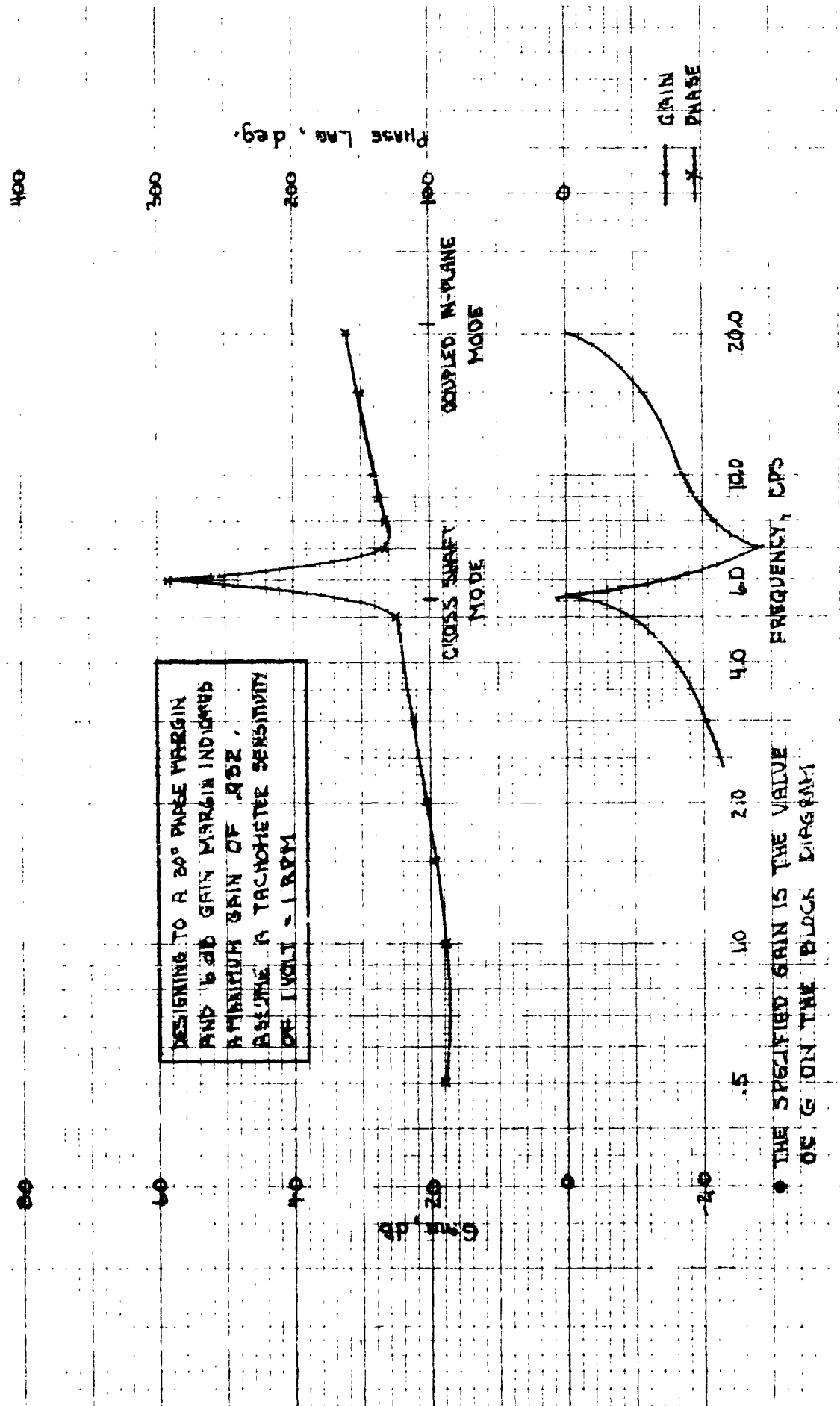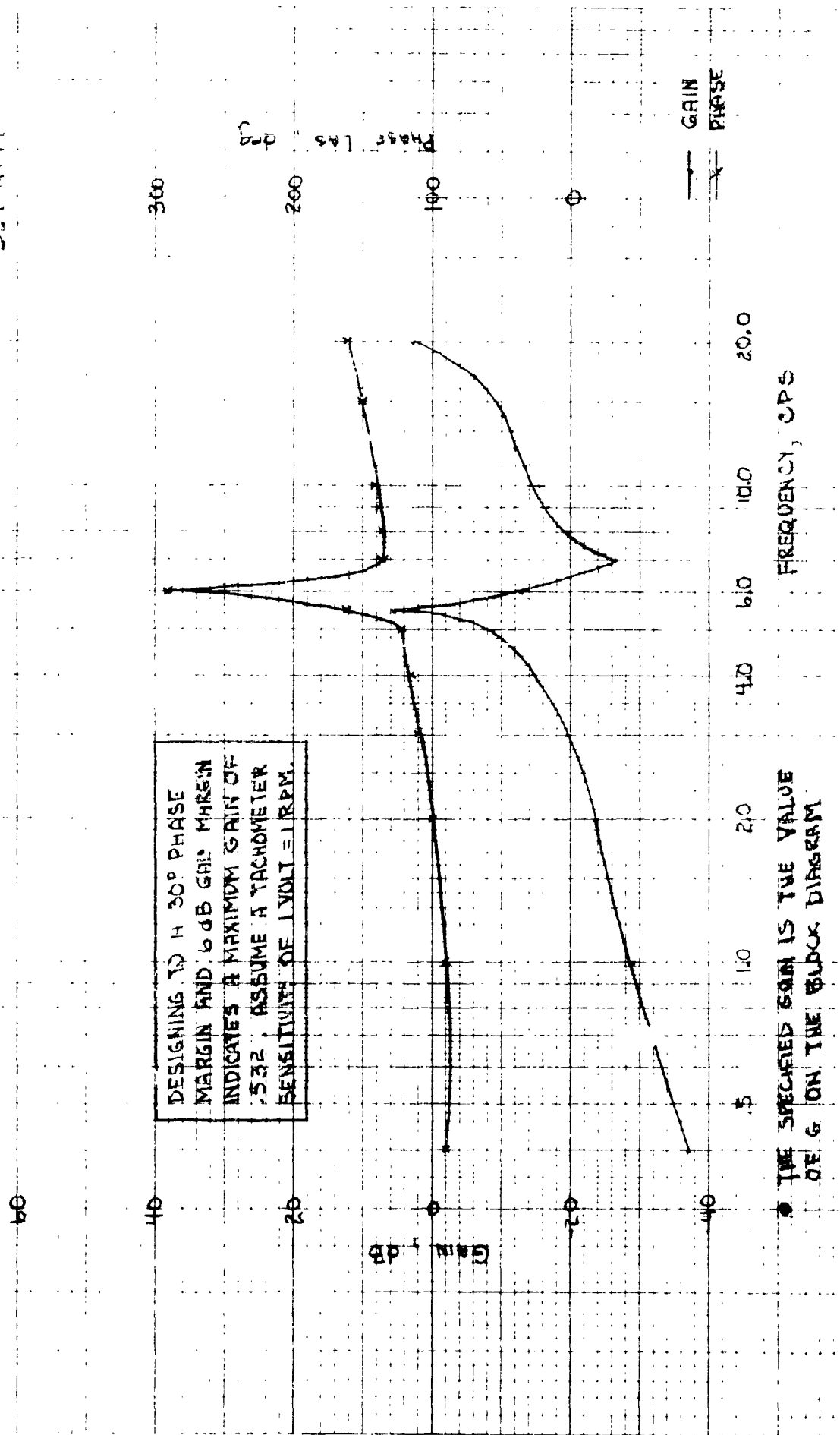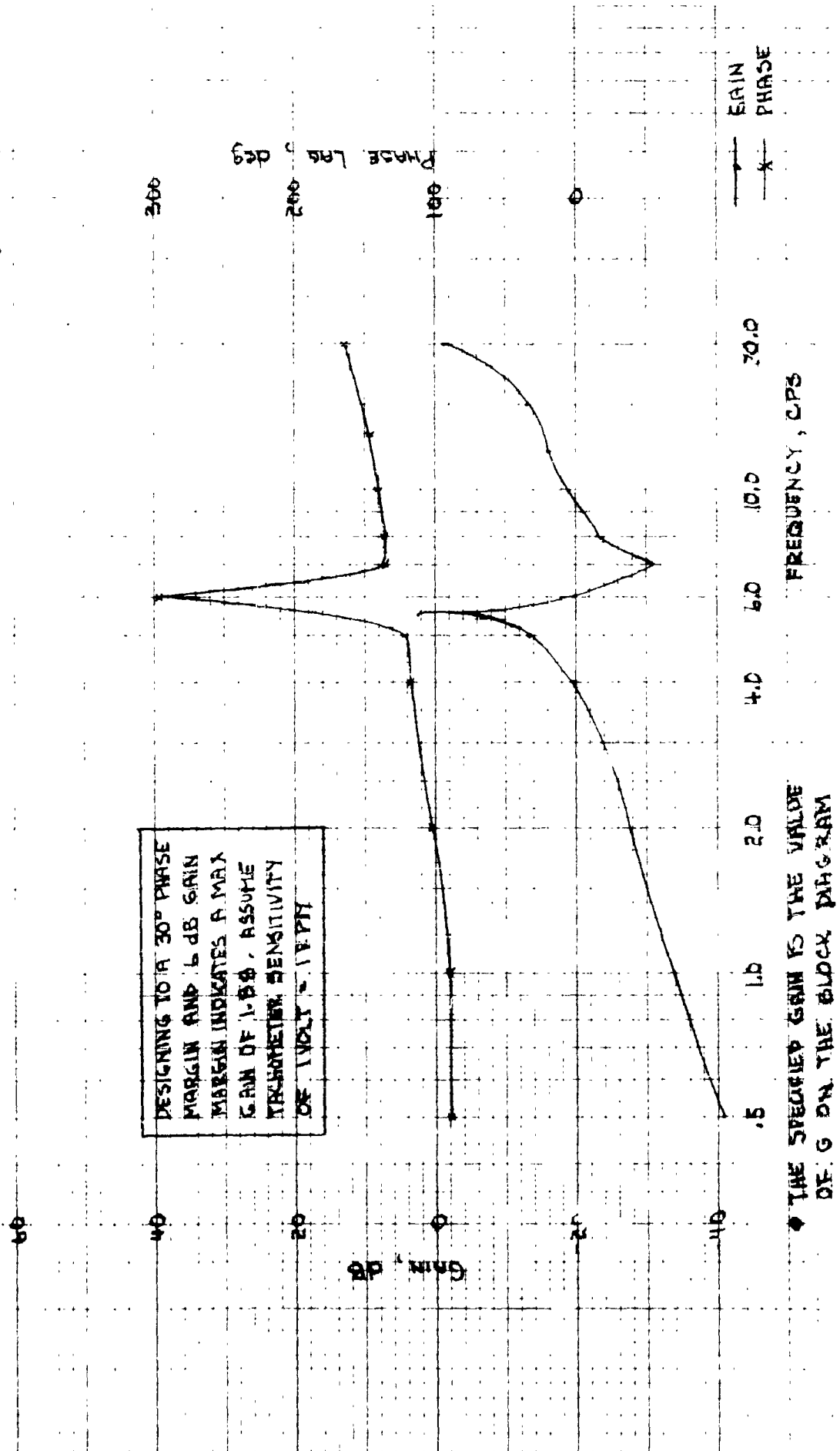DESIGNING TO 14 30° PHASE MARGIN AND 6 dB GAIN MARGIN INDICATES A MAXIMUM GAIN OF .532 . ASSUME A TACHOMETER SENSITIVITY OF 1 VOLT = 1 RPM.

* THE SPECIFIED GAIN IS THE VALUE OF G ON THE BLOCK DIAGRAM.

GAIN
PHASE

FREQUENCY, CPS

Gain , dB

Phase Lag

315

FIGURE B.6 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

HOVER CONFIGURATION
CROSS SHAFT PRESENT
20 fps DESCENT
55 RPM

DESIGNING TO A 30° PHASE
MARGIN AND 6 dB GAIN
MARGIN INDICATES A MAX
GAIN OF 1.88. ASSUME
TACHOMETER SENSITIVITY
OF 1 VOLT = 1 FPM

♦ THE SPECIFIED GAIN IS THE VALUE
OF G ON THE BLOCK DIAGRAM

— GAIN
—✕— PHASE

FIGURE B.7 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
NO CROSS SHAFT
386 RPM
100 KNOTS

DESIGNING TO A 30° PHASE MARGIN
AND 6 dB GAIN MARGIN INDICATES
A MAXIMUM GAIN OF 17.0. ASSUME
A TACHOMETER SENSITIVITY OF
1 VOLT = 1 RPM.

• THE SPECIFIED GAIN IS THE VALUE
OF G ON THE BLOCK DIAGRAM

D222-10060-3

FIGURE B.8 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
NO CROSS SHAFT
386 RPM
500 KNOTS

FIGURE B9 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

DESIGNING TO A 30° PHASE MARGIN AND 6 dB GAIN MARGIN INDICATES A MAXIMUM GAIN OF .224. ASSUME A TACHOMETER SENSITIVITY OF 1 VOLT : 1 RPM
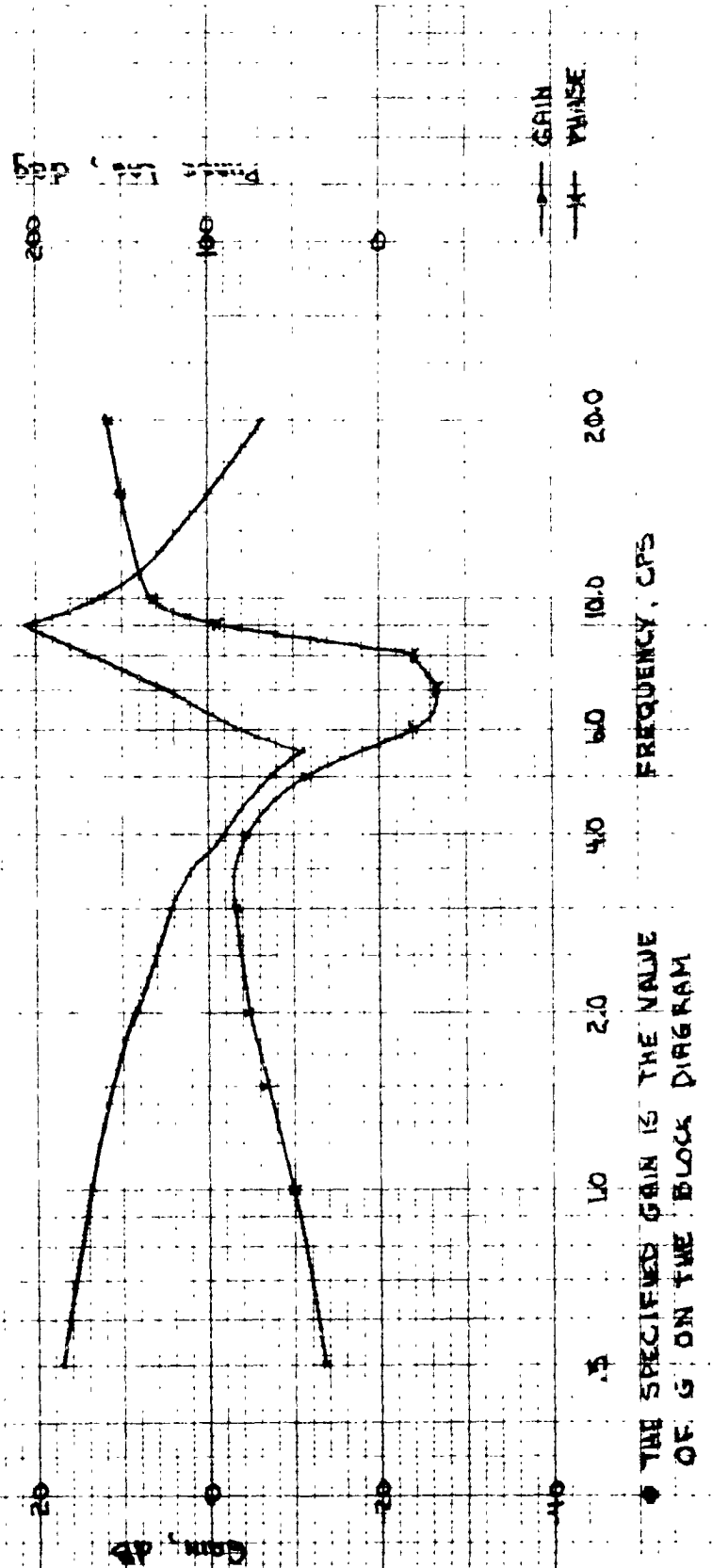
Phase Lag, deg

Gain, dB

FREQUENCY, CPS

GAIN
PHASE

♦ THE SPECIFIED GAIN IS THE VALUE OF G ON THE BLOCK DIAGRAM

CRUISE CONFIGURATION
CROSS TILT RECENT
200 KNOTS
336 RPM

FIGURE B.10 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

Phase Lag, deg

360

200

100

20.0

10.0

6.0

4.0

2.0

FREQUENCY, CPS

— GAIN
—✕— PHASE

DESIGNING TO A 30° PHASE MARGIN
AND 6 dB GAIN MARGIN INDICATES A
MAXIMUM GAIN OF .595. ASSUME
A TACHOMETER SENSITIVITY OF
1 VOLT = 1 RPM

Gain, dB
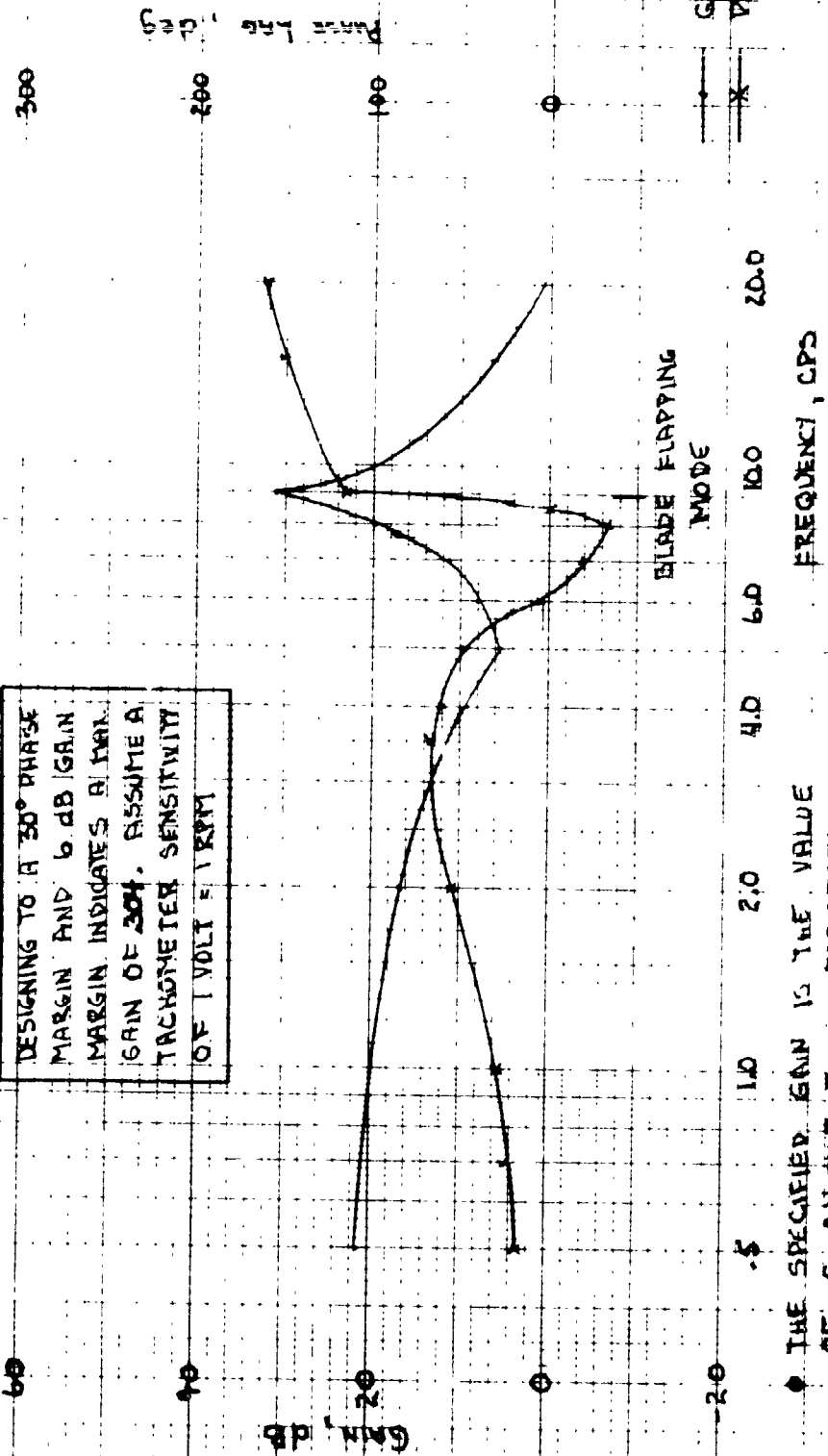
60

40

20

0

20

40

● THE SPECIFIED GAIN IS THE VALUE
OF g ON THE BLOCK DIAGRAM

320

FIGURE B.11 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
CROSS SHAFT PRESENT
200 KNOTS
436 RPM

DESIGNING TO A 30° PHASE MARGIN
A 6dB GAIN MARGIN INDICATES A
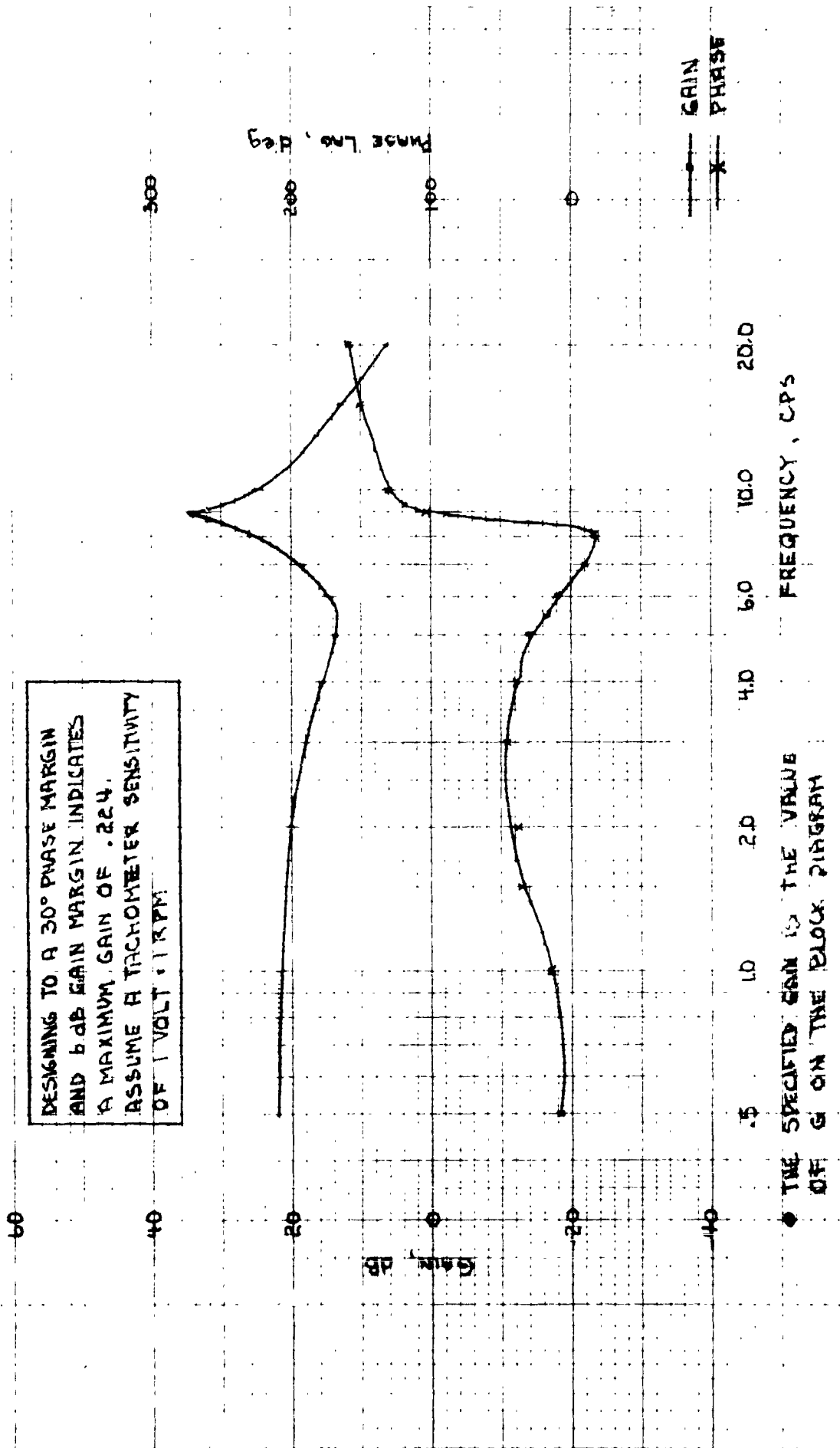MAXIMUM GAIN OF .421. ASSUME
A TACHOMETER SENSITIVITY OF
1 VOLT = 1 RPM

● THE SPECIFIED GAIN IS THE VALUE OF
G ON THE BLOCK DIAGRAM

D222-10060-3
REV. A

FIGURE B.12 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
CROSS SHAFT PRESENT
386 RPM
100 KNOTS

DESIGNING TO A 30° PHASE MARGIN AND 6 dB GAIN MARGIN INDICATES A MAXIMUM GAIN OF .232. ASSUME A TACHOMETER SENSITIVITY OF 1 VOLT: 1 RPM
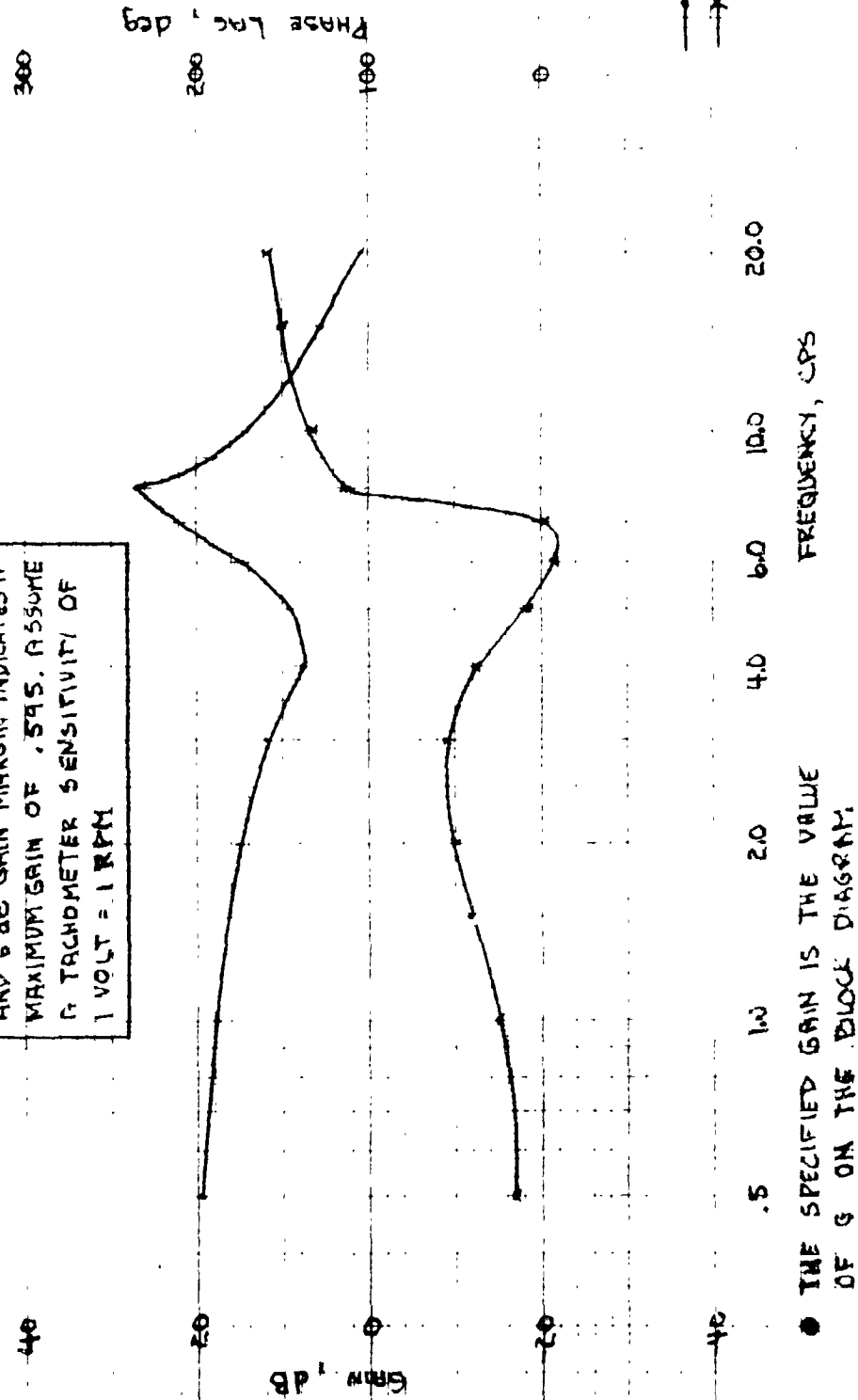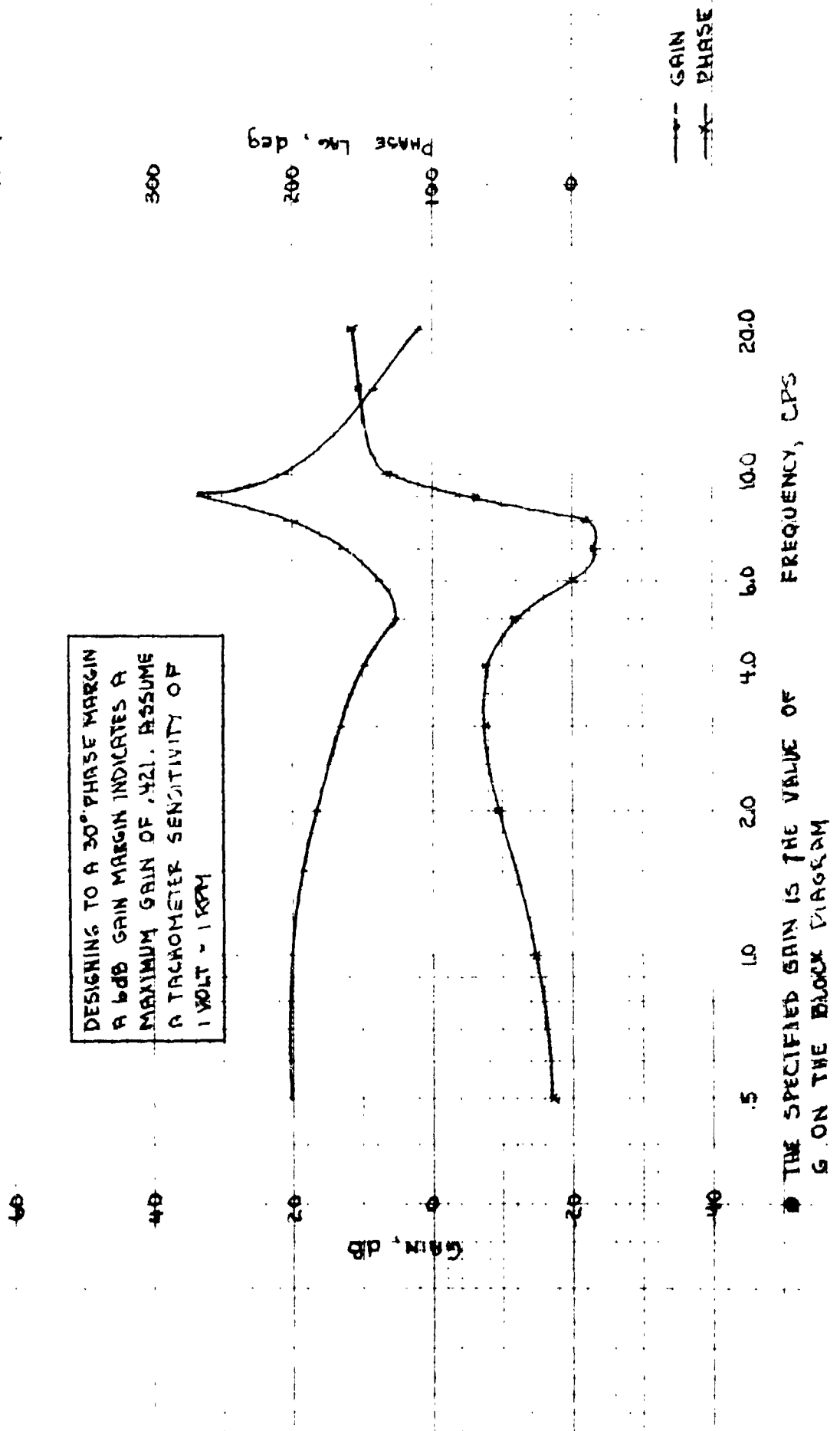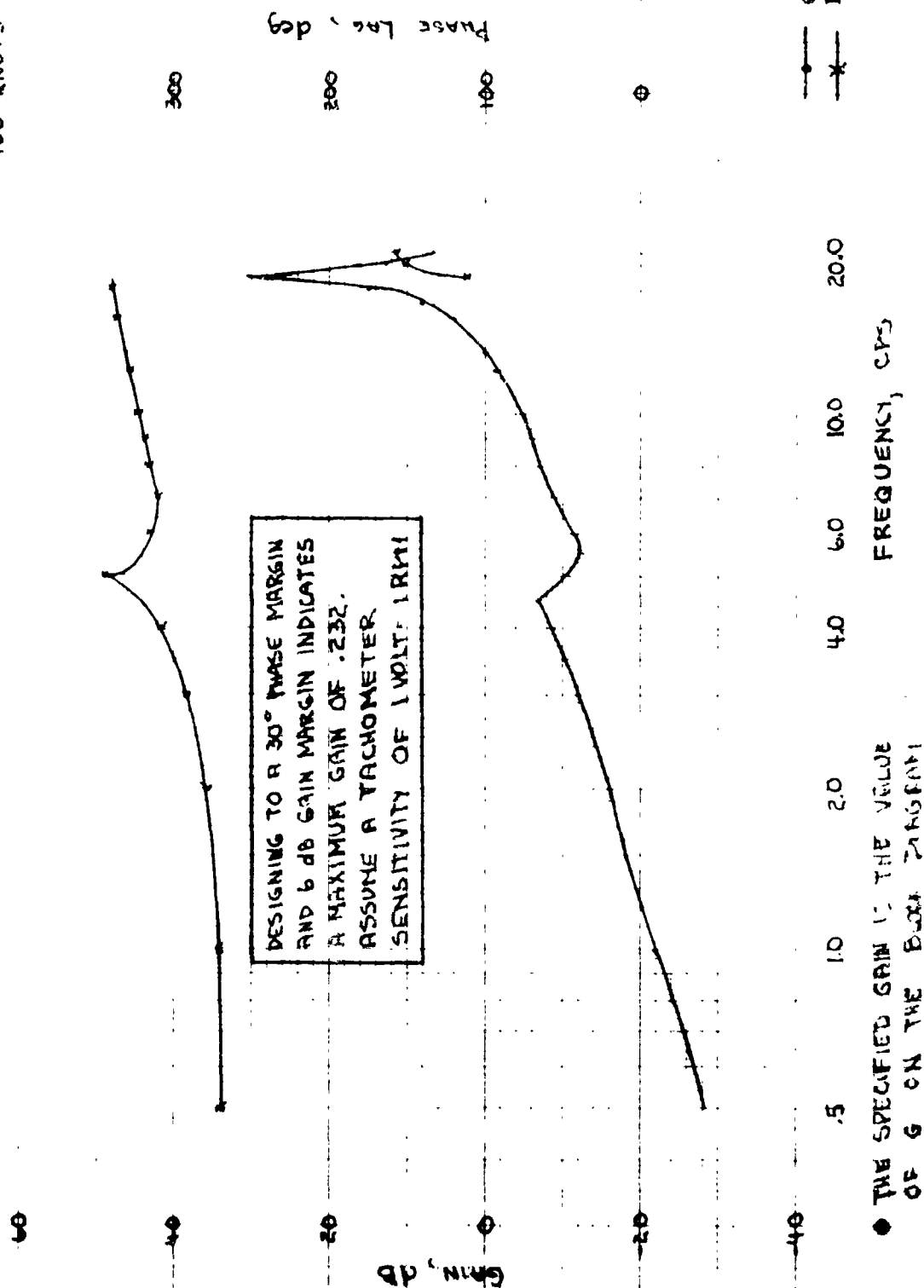
● THE SPECIFIED GAIN IS THE VALUE OF G ON THE BODE DIAGRAM
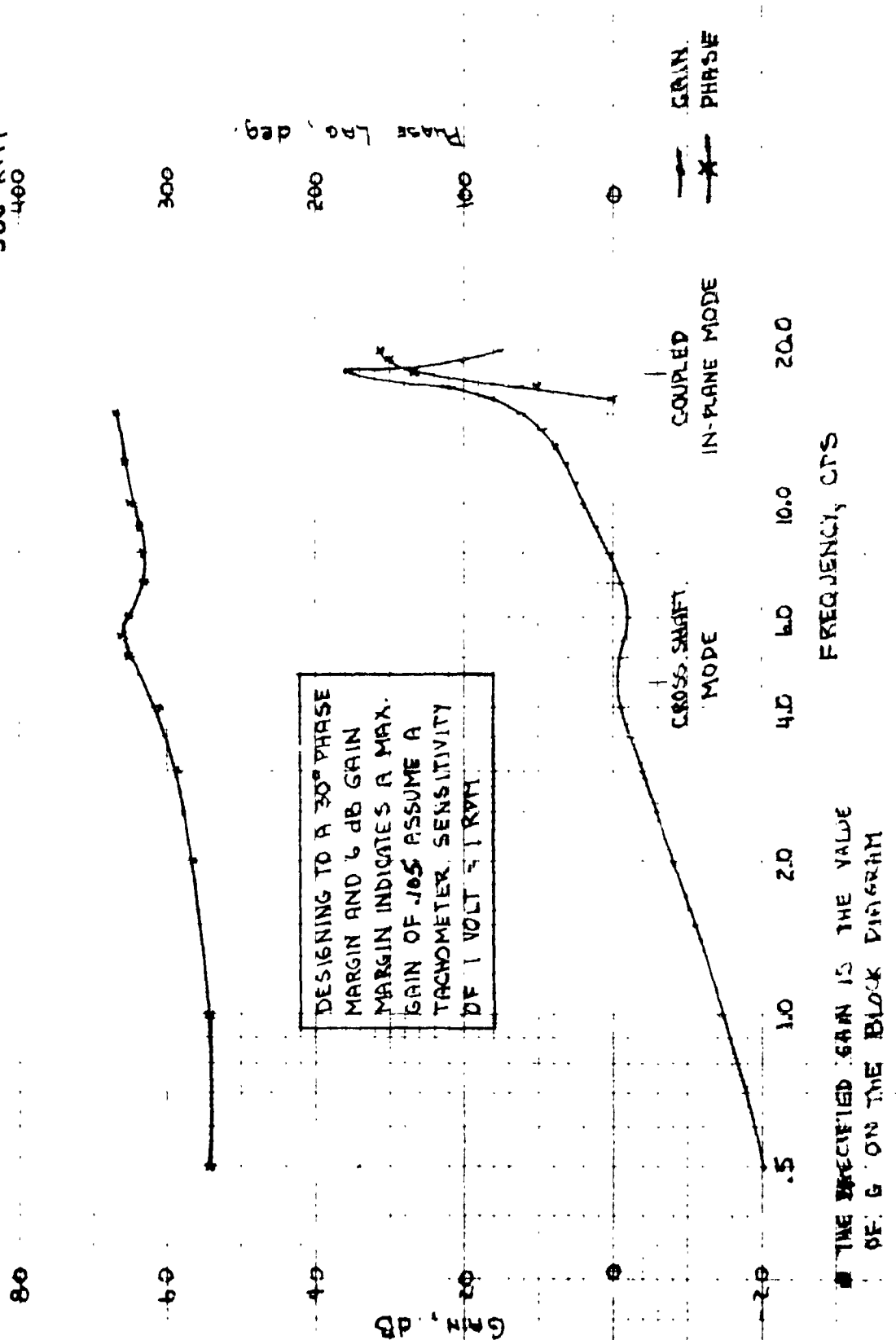
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FIGURE B-13 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
CROSS SHAFT PRESENT
386 RPM
300 KNOTS

FIGURE B.14 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

DESIGNING TO A 30° PHASE & GAIN
AND 6 dB GAIN MARGIN INDICATES
A MAX GAIN OF .0561. ASSUME
A TACHOMETER SENSITIVITY OF
1 VOLT = 1 RPM

● THE SPECIFIED GAIN IS THE VALUE
OF G ON THE BLOCK DIAGRAM

GAIN
PHASE

PHASE LAG, deg
300
200
100
0

GAIN, dB
60
40
20
0
-20
-40

FREQUENCY, CPS
.5   1.0   2.0   4.0   6.0   10.0   20.0

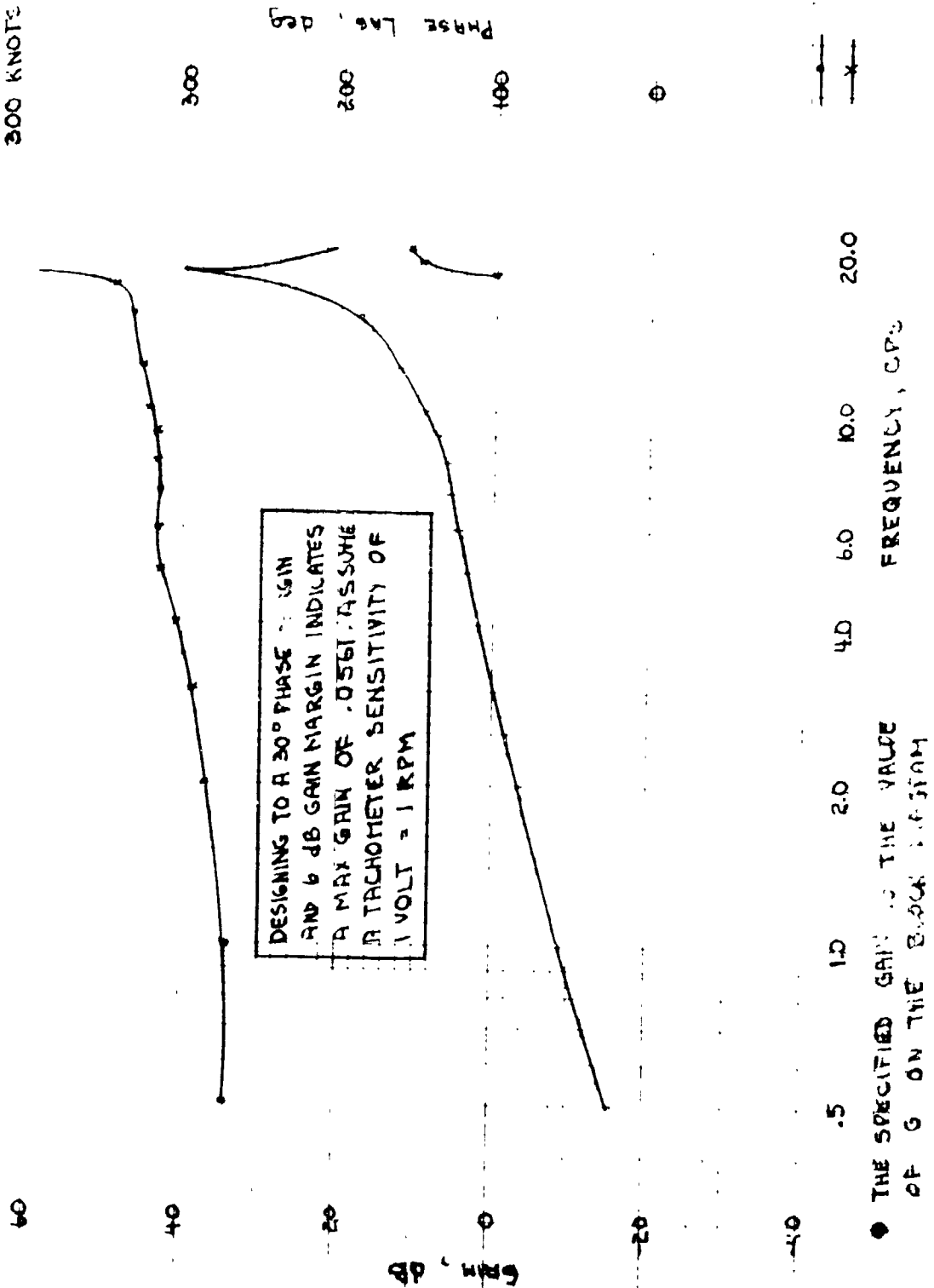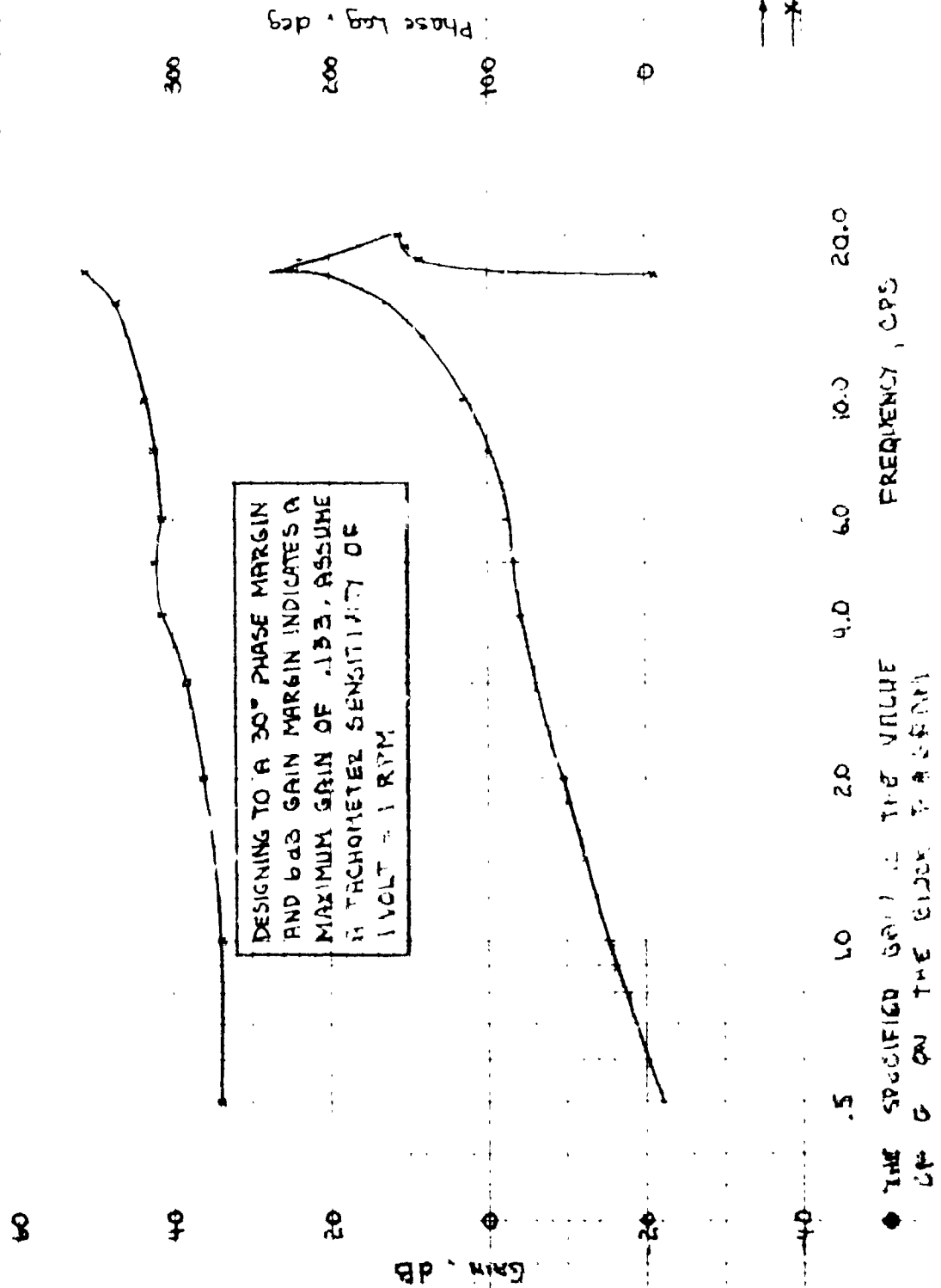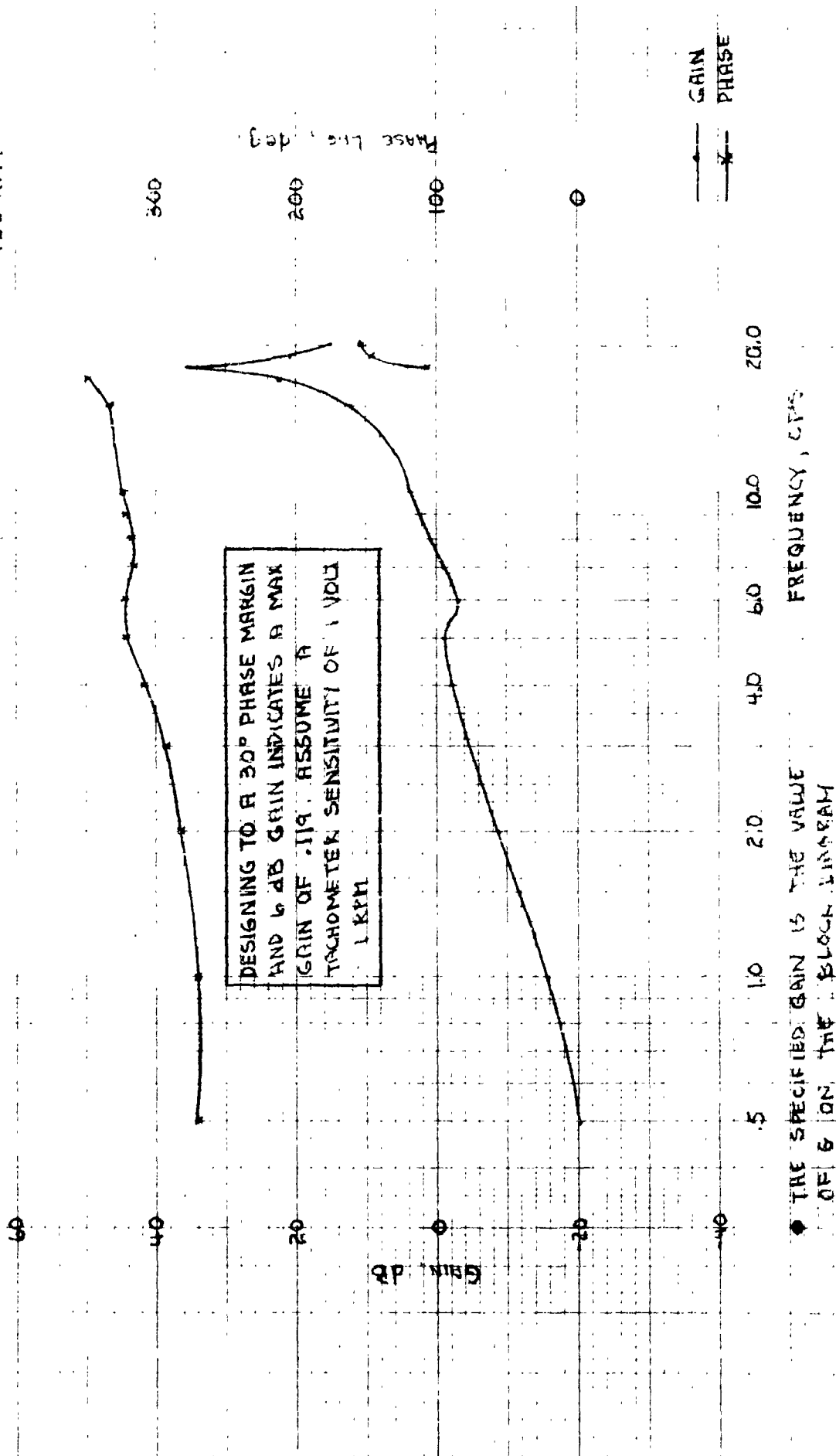FIGURE 8.15 ROTOR ROTATION FREQUENCY RESPONSE WITH ACTUATOR DYNAMICS

D222-10060-3

FIGURE B.16 ROTOR ROTATION FREQUENCY RESPONSE
WITH ACTUATOR DYNAMICS

CRUISE CONFIGURATION
CROSS SHAFT PRESENT
200 KTS
436 RPM

DESIGNING TO A 30° PHASE MARGIN
AND 6 dB GAIN INDICATES A MAX
GAIN OF .119. ASSUME A
TACHOMETER SENSITIVITY OF 1 VOLT
1 KPM

● THE SPECIFIED GAIN IS THE VALUE
OF 6 ON THE BLOCK DIAGRAM

APPENDIX C - "FLY BY WIRE": CONVERSION OF M222 CONTROL SYSTEM TO ELECTRICAL SIGNALING

Because the tilt rotor control system requires extensive mixing, gain and shaping changes as a function of flight conditions, it appears to be a good candidate for a fly-by-wire system. While a fly-by-wire study is not proposed at the present time as part of the follow on program, the following paragraphs discuss some aspects which should be given further consideration at a later date.

A block diagram for a possible fly-by-wire control system to do the same job is also given (Figure C-1). This system would use the same cockpit controls, artificial feel units and ASE actuators as the proposed Model 222. Also, the swashplate actuators, spoiler actuators and flaperon actuators would remain the same. Dual hydraulic power actuators would be added for the rudder and for the or. Thirteen quadruple-driven actuators would then be added as shown to convert electrical signal to mechanical inputs at the hydraulic power actuators. These could possibly be electro-ram type actuators of the type we now propose use or the governor system on the Model 222 (or they could be electro-hydraulic).

A summary of the actuator tradeoff is given below. The rest of the tradeoff consists of replacing five ratio changers, eight scheduling cams, fourteen stages of mechanical mixing,

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THE *BOEING* COMPANY

TILT ROTOR FLY-BY-WIRE CONTROL SYSTEM

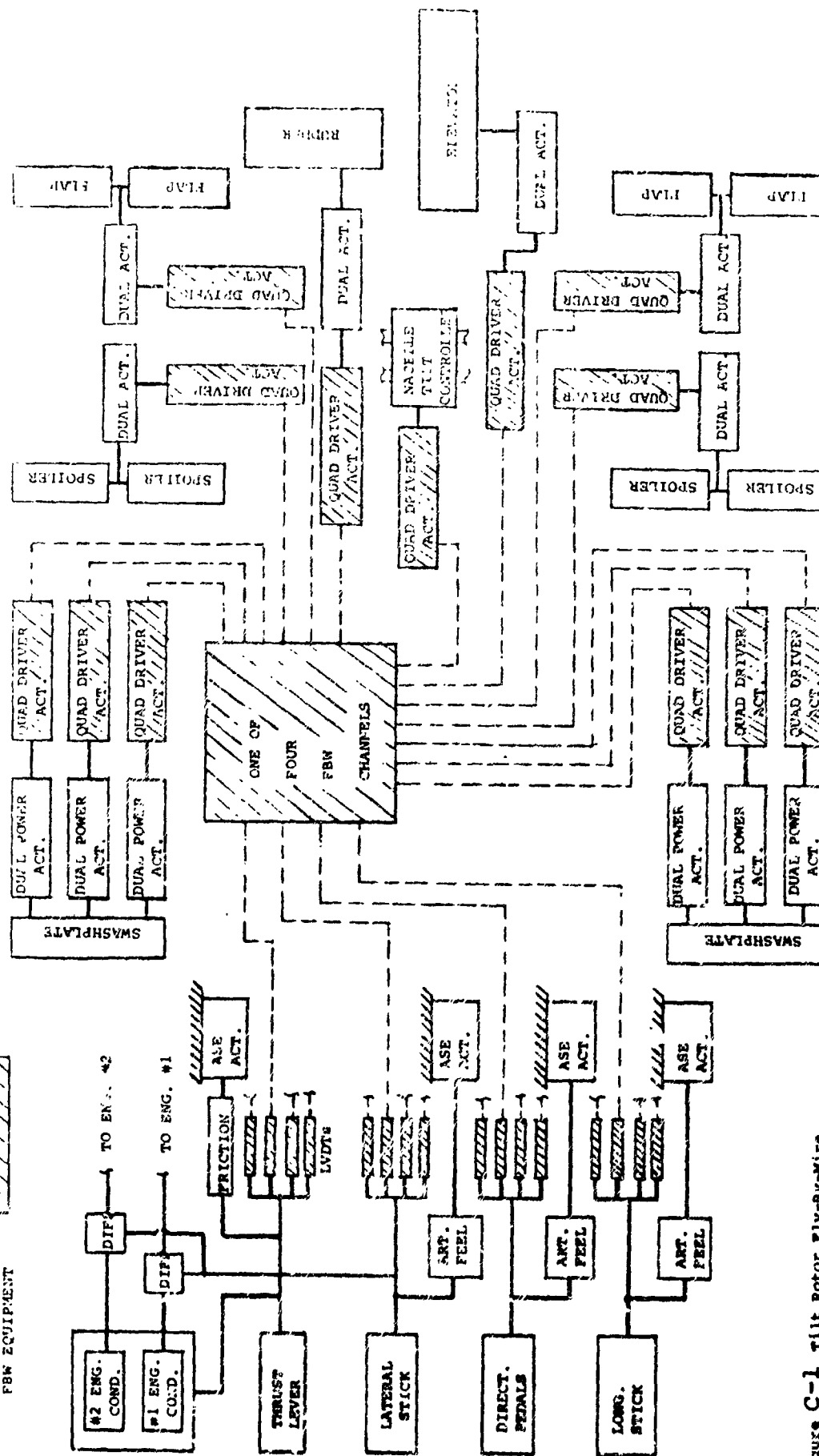SHADED BOX INDICATES SPECIAL FBW EQUIPMENT

Figure C-1 Tilt Rotor Fly-By-Wire Control System (FBW)

(mechanical differentials), four governor speed settings LVDTs,
four collective pitch "quickening" LVDTs, and 90% of the
mechanical routing linkage -- with, sixteen cockpit LVDT pick-
offs, four individual channel FBW control boxes and the asso-
ciated wiring. It is assumed that the sensors required and
much of the automatic system electronics would be the same in
both systems. The tradeoff in actuators between the tilt
rotor FBW control system and mechanical system with full load
alleviation/mode suppression capability is:

DELTA ACTUATOR COUNT - remove from mechanical system

     a)   4 dual stick boost actuators

     b)   8 dual extensible link SAS actuators

     c)   1 triple extensible link SAS actuator

     d)   1 quadruple electromechanical governor actuator

     e)   1 electromechanical DCP trim actuator

ADD FOR FBW SYSTEM

     a)   13 quadruple-driven actuators

     b)   2 dual power actuators at rudder and elevator